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Lone Star Aerospace Presents:

F L A R E

The Far Side Lunar Astronomy Research Expedition - A Design of a Far Side Lunar Observatory

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Executive Overview

This document outlines the design completed by members of Lone Star Aerospace, Inc. (L.S.A.) of a lunar observatory on the far side of the Moon. Such a base would not only establish a long term human presence on the Moon, but would also allow more accurate astronomical data to be obtained.

A lunar observatory is more desirable than an Earth based observatory for the following reasons:

- Instrument weight is reduced due to the Moon's weaker gravity.
- Near vacuum conditions exist on the Moon.
- The Moon has slow rotation to reveal the entire sky.
- The lunar surface is stable for long baseline instruments.

All the conditions listed above are favorable for astronomical data recording.

The technical aspects investigated in the completion of this project included site selection, mission scenario, scientific instruments, communication and power systems, habitation and transportation, cargo spacecraft design, thermal systems, robotic systems, and trajectory analysis.

The site selection group focused its efforts on finding a suitable location for the observatory. Hertzsprung, a large equatorial crater on the eastern limb, was chosen as the base site.

Primary and Secondary Base Designs:

Two possible base designs were developed. After analyzing these two designs, a primary base design and a secondary base design were selected. These two designs differ in the positioning of the larger instrument packages that will be placed on the lunar surface as well as in the type of habitat module that will be utilized. The primary base design consists of a main base with a Space Station Common Module (SSCM) type habitat and three large independent instrumentation fields - one separate field for the Very Low Frequency Array (VLFA), one for the Optical Interferometer (OI), and one for the Submillimeter Interferometer (SI). The secondary base, on the other hand, consists of a main base with an inflatable habitat and one large instrument field in which the fields for the VLFA, OI, and SI overlap each other.

The advantages of the primary base were analyzed. The main advantages of this base were as follows:

- Less interference between elements of the VLFA, OI, and SI
- Easier placement and maintenance of the habitat
- Easier expansion of any of the large instrumentation fields
- Easier maintenance of an instrument element (since maintenance would not cause dust build up on nearby instruments as it would in the secondary base's overlapping instrumentation field).

The advantages of the secondary base were as follows:

- Less range required by transportation and robotics elements
- Larger habitat
- Less power and communications cable required to reach the instruments.

After analyzing these advantages and considering the fact that the main purpose of constructing the base is to obtain the most accurate astronomical data possible, the base with the SSCM and three independent instrumentation fields was chosen as the primary base. A sketch of the primary base is shown in Figure 1.

Overview of Subsystems:

The design of the far side lunar observatory involved investigation into seven subsystems. These subsystems included instrumentation, habitation and transportation, power and communications, robotics, thermal systems, cargo spacecraft design, and trajectory analysis. The following sections give a brief overview of each of these subsystems.

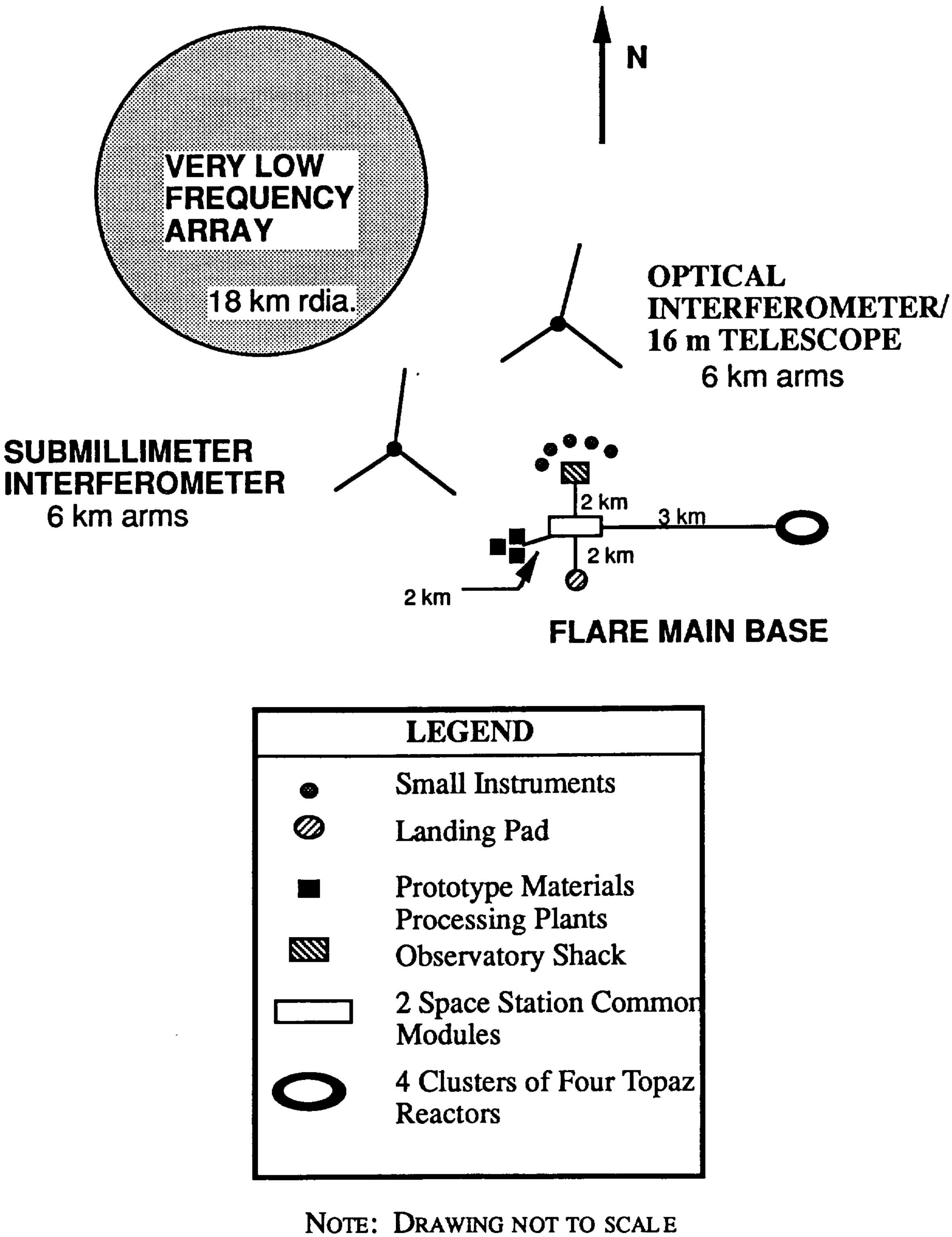

Figure 1. Primary Base Configuration.

Instrumentation. Astronomical, geological, and environmental instrumentation packages will be emplaced on the lunar surface. The following is a list of the major instruments to be utilized:

- Very Low Frequency Array
- Submillimeter Interferometer
- Optical Interferometer
- Transit Telescope
- 16 m Telescope
- Moon-Earth Radio Interferometer.

The mass of the total instruments package has been calculated to be 91 metric tons.

Habitation and Transportation. Two SSCM modules connected end to end will provide for habitation on the lunar base. Two airlocks at either end of this arrangement will provide adequate ingress and egress. A partially-closed environmental control and life support system will be utilized. MOSAP and LOTRAN vehicles will provide lunar transportation.

Communication and Power. During the construction phase, a satellite in an L2 halo orbit will relay data from the lunar surface to a geostationary satellite in Earth orbit to the Earth's surface. When the base becomes fully operational, however, a radio-free sky is desired to take accurate astronomical readings. Therefore, a fiber optic cable will be used as a communication link from the base to a transmitter/receiver station on the near side of the Moon. It will be laid out by a robotic rover from the base to the limb of the Moon. From there, the signal can be broadcasted directly to Earth without interfering with astronomical observations.

The base will be powered by four clusters of four Soviet manufactured Topaz reactors. These will supply the base with approximately 160 kWe of energy. Use of this cluster arrangement will prevent the total loss of power to the base in the event of a failure. If an emergency occurs and a cluster must be shut down, the other reactors can still produce 120 kWe for the base.

Robotics. Four robotic elements will set up the far side lunar base. They include a crane, an excavator/digger, and two assembly robots. They will dig holes, bury the habitation modules and reactors, lay power and communications cable, and set up the instruments. These robotic elements will use a combination of artificial intelligence and tele-robotics to successfully navigate and construct the base.

Thermal Systems. The lunar base will be thermally controlled with the use of both radiators and heat exchangers. Radiators will be used to cool the reactors and heat exchangers will be used to cool the habitat and some of the smaller astronomical instrument packages. Manufactured shades will be used if passive cooling of the larger instrument packages is necessary.

Cargo Spacecraft Design. A cargo spacecraft designed by Eagle Engineering will be used to carry the 180 metric tons of materials from Low Earth Orbit to Low Lunar Orbit. A Lunar Operations Vehicle will then transport these materials to the lunar surface.

Trajectory Analysis. Cargo spacecraft trajectories will consist of "spiral-in" type trajectory with a time of flight of approximately 130 days. Any manned missions to the base will use hybrid free-return trajectories.

Management and Cost:

Lone Star Aerospace is composed of a project leader, integration leader, chief technical engineer, administrative leader, and seven technical departments (each with its own department leader). This type of management structure has worked quite efficiently. No major problems have arisen in the design of the far side lunar observatory.

A cost analysis on the design of the lunar base has been performed based on the hardware costs incurred over the past fifteen weeks as well as the number of man-hours utilized. These figures were then compared to the estimated cost for the project as presented in the proposal. The total cost for the design of this base has been calculated to be $51,853., well under the budget agreed upon in the proposal.

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List of Acronyms

AI	Artificial Intelligence
AT&T	American Telephone and Telegraph
ATCS	Active Thermal Control System
ATDRSS	Advanced Tracking and Data Relay Satellite System
CARD	Computer Aided Remote Driving
CCD	Charge-Coupled Diode
CDR	Critical Design Review
DARPA	Defense Advanced Research Projects Agency
DOE	Department of Energy
DRS	Data Relay System
ECLSS	Environmental Control and Life Support System
FLARE	Far Side Lunar Astronomical Research Expedition
GCR	Galactic Cosmic Radiation
GRT	Gamma Ray Telescope
HECRD	High Energy Cosmic Ray Detector
HLC	Heavy Lift Crane
HST	Hubble Space Telescope
IM	Inflatable Module
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOTRAN	Local Transportation Vehicle
LOV	Lunar Operations Vehicle
L.S.A.	Lone Star Aerospace
LTT	Lunar Transit Telescope
MERI	Moon-Earth Radio Interferometer
MOSAP	Mobile Surface Application Traverse Vehicle
MT	Metric Ton
NASA	National Aeronautics and Space Administration
NEP	Nuclear Electric Propulsion
NSO	Nuclear Safe Orbit
NSTS	National Space Transportation System
OI	Optical Interferometer
OTV	Orbital Transfer Vehicle

PDR	Preliminary Design Review
PTCS	Passive Thermal Control System
REM	Roentgen Equivalent Man
RFP	Request for Proposal
RTG	Radioisotope Thermal Generator
SAM	Semi-Autonomous Mobility
SI	Submillimeter Interferometer
SPE	Solar Particle Event
SSCM	Space Station Common Module
TCS	Thermal Control System
USRA	Universities Space Research Association
VLFA	Very Low Frequency Array

1.0 General Summary

This document presents the design of a far side lunar observatory by the Lone Star Aerospace design team. This design began in response to the Request For Proposal (RFP) #F91/274L issued by the National Aeronautics and Space Administration (NASA) in conjunction with the Universities Space Research Association (USRA) [1]. An overview of the mission design, primary and secondary base layouts, and a presentation of subsystems involved in this design are discussed. In addition, the management structure and estimated cost involved in completing this design are presented.

1.1 Project Background

An initiative proposed by NASA in 1987 suggested the establishment of a far side lunar observatory to expand human presence in space. Placing an observatory on the far side of the Moon has several advantages and disadvantages. They are detailed in sections 1.1.1 and 1.1.2 below.

1.1.1 Lunar Astronomy Advantages

Performing astronomical observations from the lunar surface provides many advantages over both Earth-based and Earth-orbiting astronomy. Some of these advantages include the following:

- Existence of a near vacuum and dark sky on the Moon
- Size and stability of the lunar surface for long baseline instruments
- Partial cosmic ray protection
- Near-cryogenic environment
- Low gravity to reduce instrument support mass
- Slow rotation to reveal the entire sky
- Shielding from Earth and its electromagnetic environment.

1.1.2 Lunar Astronomy Disadvantages

Performing astronomical observations from the lunar surface also has some drawbacks which must be addressed. These limitations include the following:

- Constant presence of cosmic radiation
- Risk of micrometeoroid impact
- Large thermal variations
- Logistics of placing the base on lunar surface
- Effects of lunar dust on instruments
- High cost of a large scale mission.

1.2 Assumptions

In order to narrow the scope of this project, Lone Star Aerospace made the following assumptions:

- All payloads needed for construction of the lunar observatory will be transported from Low Earth Orbit (LEO).
- Precursory missions to determine an appropriate site for the observatory have already been completed.
- Robotic capability exists to allow for construction of the observatory with minimal human assistance.
- Ion propulsion is a feasible means by which to transport construction materials and instruments from LEO to the Moon.

1.3 Design Requirements

The following section discusses the groundrules established by Lone Star Aerospace in the design of the far side lunar observatory. These groundrules were adapted from NASA's original guidelines:

- A period of five to six years will be allowed for base construction with the initial mission beginning in the year 2005.
- Occasional manned missions may be implemented to provide any necessary maintenance during the construction phase.
- There will be no more than one mission per year so as to not overburden the National Space Transportation System (NSTS).

- The manned missions will consist of four to six crew members with a maximum lunar stay of 14 Earth days.
- Constant communication with the Earth is required during both construction and operation of the lunar base.
- A base location near the lunar equator on the far side of the Moon is desirable to maximize the field of view and limit any horizon interference from the Earth.

1.4 Mission Scenario

The following table outlines the proposed mission scenario for the construction of the FLARE base. At least one manned mission will occur in parallel with the third construction mission to confirm the ability of the base to sustain crew members.

Table 1.1. Mission Scenario.

Date	Mission Type	Mission Focus
2005	Construction	Robotics, Power, and Astronomical Instruments
2006	Construction	Habitat and Instruments
2007	Construction	Astronomical Instruments
2007	Manned	Verification of Construction, Feasibility of Manned Stay
2008	Construction	Instruments
2009	Construction	Two 4 m Telescopes
2010	Construction	Completion of Base

2.0 Primary and Secondary Base Layouts

Two base layouts have been designed. They differ in the placement of the instruments packages and the types of habitat modules used. Figures 2.1, 2.2, and 2.3 show the FLARE base alpha configuration, a close-up of the main base, and beta configuration, respectively.

Note that the launch pad is placed to the south of the entire base for both base configurations. This is to avoid endangering the habitat, power facilities, or instrument packages in the event that a lander overshoots the landing pad on its equatorial approach. Both base designs also include material processing plants. These manufacturing plants are not meant to be full scale manufacturing plants, rather, they will be prototypes that will mine and manufacture materials to investigate the feasibility of having large scale plants to support future lunar bases.

In the alpha configuration, the very Low Frequency Array (VLFA) as well as the submillimeter and optical interferometers are placed independently of one another. Furthermore, a Space Station Common Module (SSCM) type habitat is utilized in this configuration.

The alpha configuration has some obvious advantages and disadvantages. Placement of the three large instrument fields independently of one another should reduce the interference between them and allow for more accurate astronomical readings. Expansion of the alpha configuration's large astronomical fields should be fairly easy since the fields are set apart from one another. However, this base does have certain disadvantages. Placement of these three fields so far from the main base would require the transportation and robotic elements to have a fairly large range. Also, longer power cable lines would be needed to reach these instruments.

The beta configuration also has advantages and disadvantages. In the beta configuration, the three large astronomical fields overlap each other and are closer to the main base. Placing the three large astronomical fields close to one another and the main base would cut down on the range required by the transportation and robotic elements and on the length of power cable required. Furthermore, a larger habitat is used in this configuration. However, the overlapping of the three large instrumentation fields may increase the interference between the large astronomical packages, increase the difficulty involved in expanding any of these fields, and cause a variety of maintenance problems.

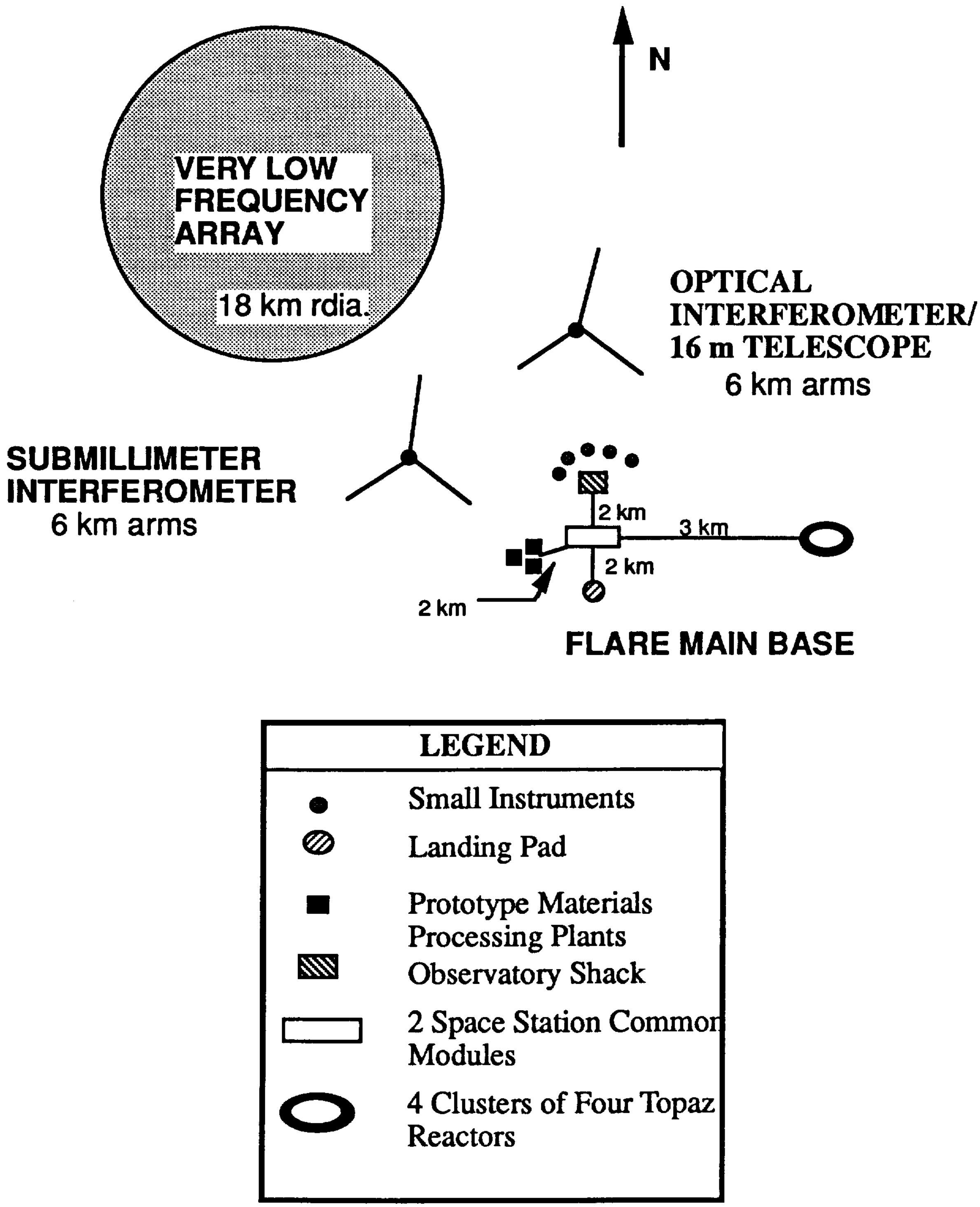

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Figure 2.1. Alpha Configuration.

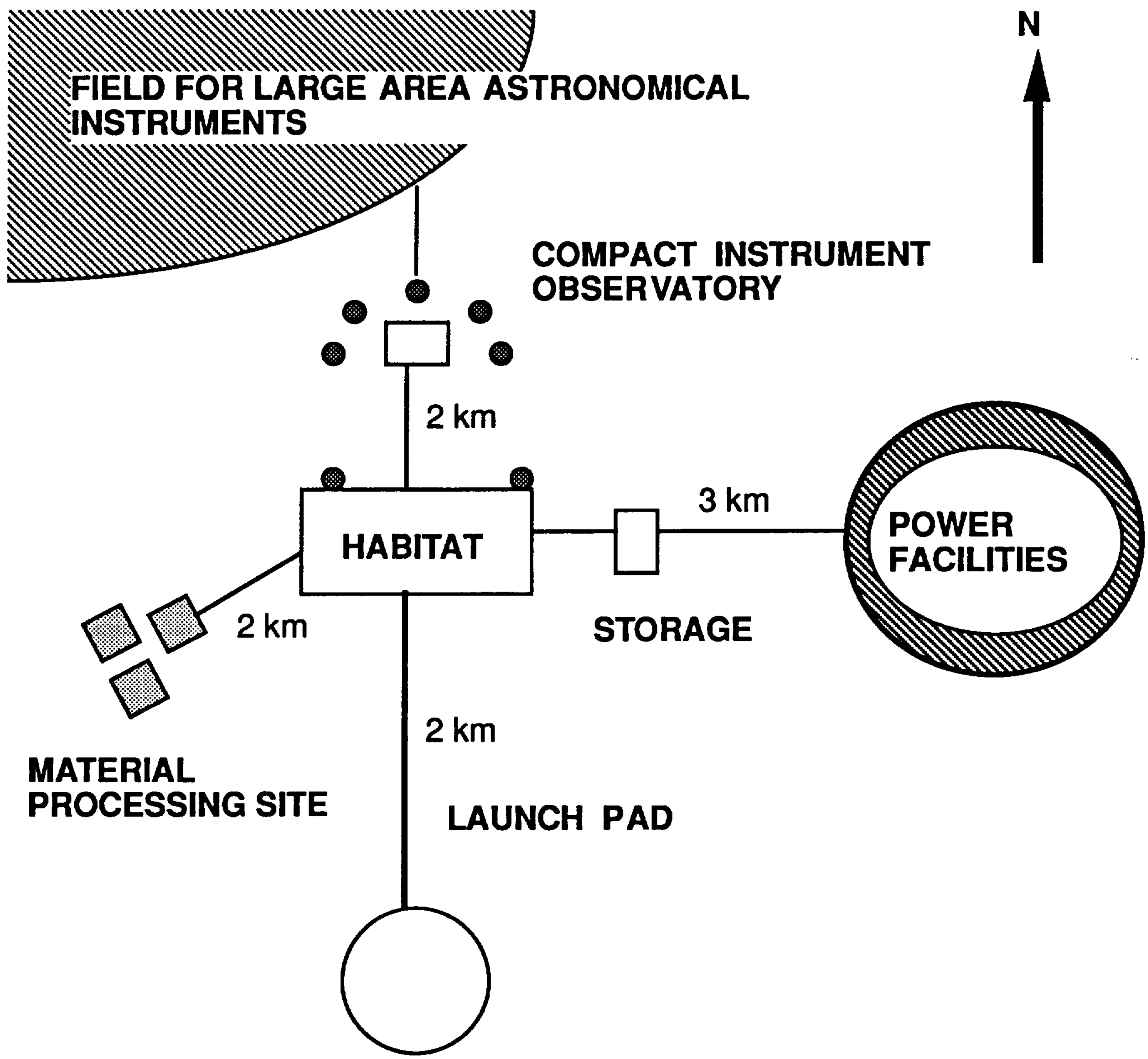


Figure 2.2. Main Base Close-Up.

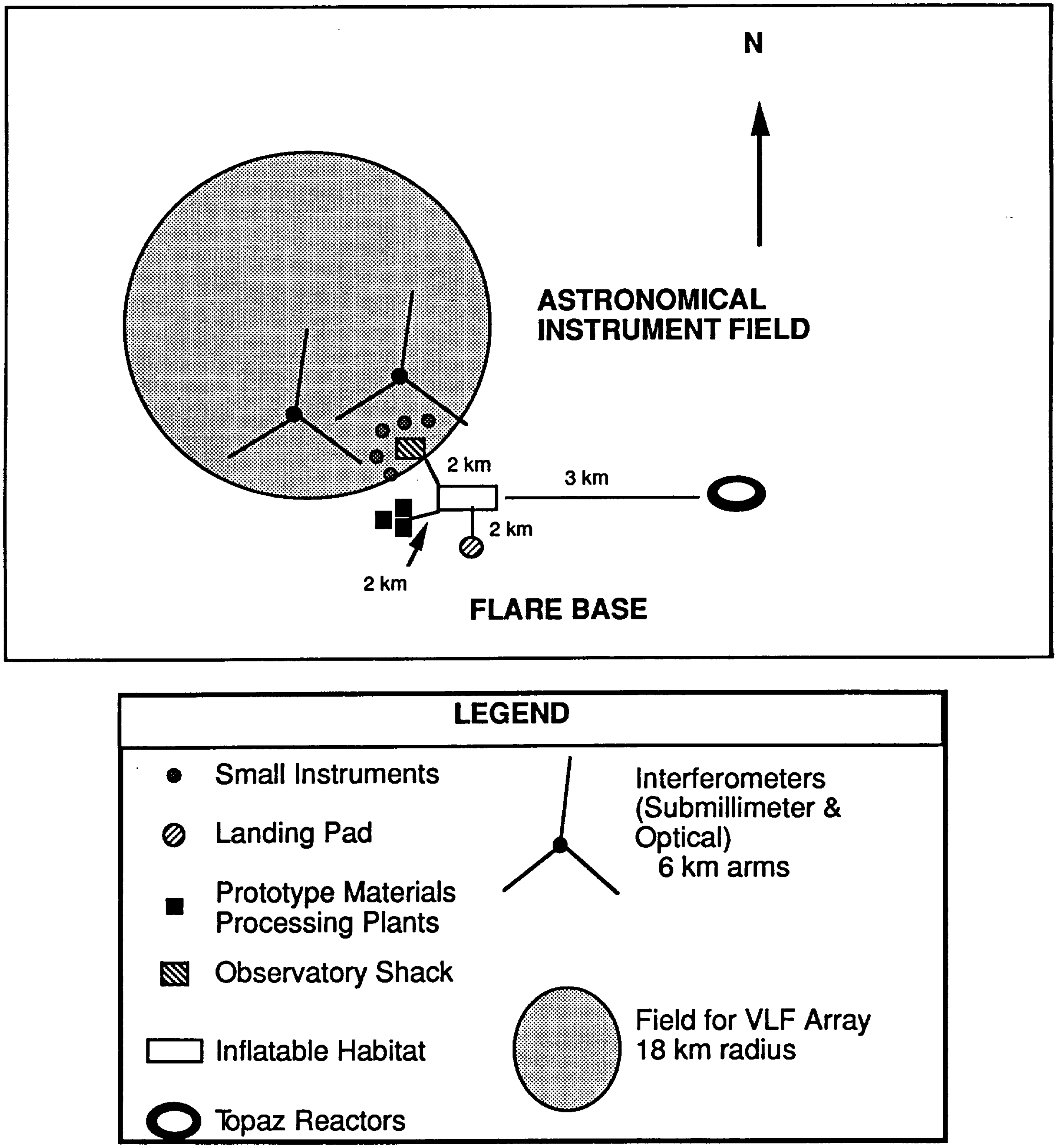


Figure 2.3. Beta Configuration.

2.1 Selection of the Primary Base Design

The advantages and disadvantages of both the alpha and beta base designs were analyzed. The alpha base configuration was chosen as the primary base since it will best suit the needs of the FLARE mission. A decision matrix showing how the primary and secondary base design were chosen is shown in Table 2.1. Each base was given a score from one to ten in the categories shown (Note: 10 is best, 1 is worst). These scores were then multiplied by a weighting factor characterizing the level of importance of the criterion being described.

Table 2.1. Decision Matrix for Choosing the Primary and Secondary Base Designs.

Factor	Weight	Alpha Base	Beta Base
Maintenance	3	7	5
Expansion	4	8	4
Low Interference with Other Fields	5	9	5
Small Range Requirements	3	4	8
Small Cable Requirements	2	6	8
Total		**122**	**96**

2.2 Site Selection

The Lone Star Aerospace design team investigated several sites as possible base locations and has selected the lunar far side crater Hertzsprung as the location of the FLARE base and observatory. Hertzsprung is a multi-ring basin located at coordinates 0° North and 130° West [1].

2.2.1 Criteria and Selection

The first criterion considered in the selection of a base site was the existence of a sufficiently large, level area for the scattered, large scale instruments that would be placed on the FLARE base. This criterion located five possible base locations at latitudes less than 35° from the lunar equator. These sites included the craters Mendeleev, Korolev, Hertzsprung, Gagarin, and Mare Moscoviense. It was decided that a base location within 20° of the lunar equator would be preferred as this would allow for simpler Earth-Moon trajectories [2]. This reduced the number of potential sites from five to three: Mendeleev at 5.6° North and 141.5° East, Korolev at 4.4° North and 157.4° West, and Hertzsprung at 0.0° North and 130.0° West [3].

These three potential sites were then examined using surface maps of the lunar far side. When compared to Mendeleev and Hertzsprung, Korolev was found to be badly eroded and filled with numerous smaller craters. Thus, Korolev was eliminated as a potential site for the FLARE observatory [4].

The lunar surface at longitudes less than 82° East or West is permanently oriented towards the Earth [2]. Mendeleev is 1,800 km from 82° East. Hertzsprung is slightly closer at 1,400 km from 82° West. Furthermore, examination of lunar far side charts revealed that Mendeleev has several deep canyons crossing its crater floor and is uniformly covered with relatively significant secondary craters. These may interfere with the set up and utilization of ground communication systems. Hertzsprung, however, was seen to have its northeast quadrant relatively free of large secondary craters [1]. For these reasons, Hertzsprung has been chosen to be the location at which Lone Star Aerospace will construct its observatory. However, Mendeleev will be retained as an alternative site.

2.2.2 Background Information on Hertzsprung

Hertzsprung is an ancient multi-ring basin. The outer ring measures 570 km in diameter [3] and the inner ring measures 260 km in diameter [1]. Lunar orbiter photographs of Hertzsprung have been obtained. In Figure 2.4, Hertzsprung is shown with both its inner and outer rings, the northeast quadrant, and several typical secondary craters between 10 and 30 km wide [4]. The northeast quadrant of the inner ring will be the location of the FLARE observatory.

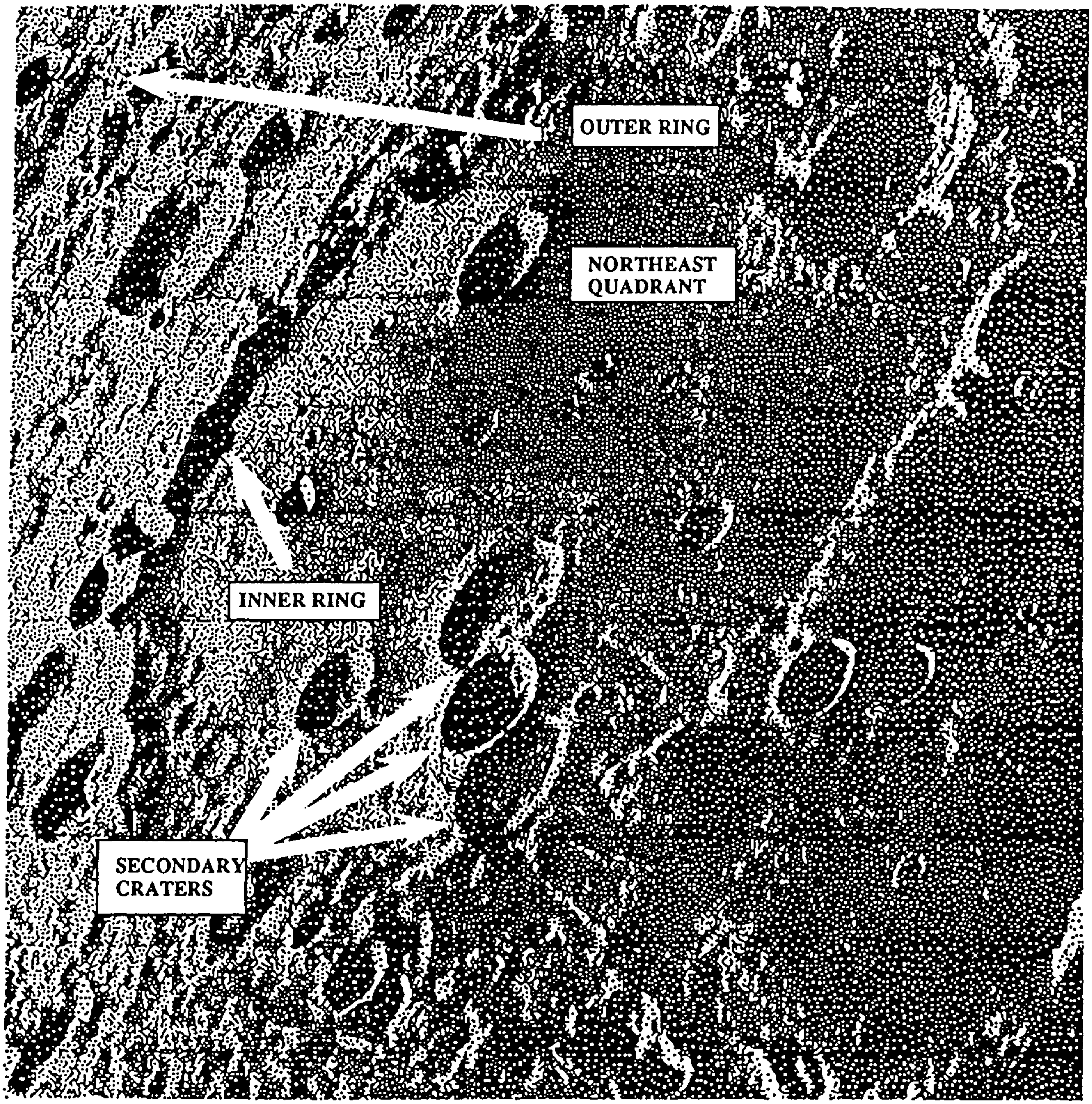


Figure 2.4. Hertzsprung [1].

2.3 Precursory Missions

Current data on the lunar far side is extremely limited. Specific data on craters such as Hertzsprung must be interpreted either from old photographs which are twenty or more years of age or from information obtained by planetary probes [5]. To put it simply, there is little data on specific areas of the lunar far side. Before an astronomical observatory may be placed on the lunar surface, more detailed knowledge and analysis is required. It is assumed that this data will be obtained both from a Lunar Orbiter and a multi-vehicle reconnaissance and beacon emplacement precursory mission.

2.3.1 Precursory Mission 1: The Lunar Orbiter

NASA currently has the Lunar Orbiter mission planned for either the middle or late portion of this decade. Its primary goals will be to gather geochemical, mineralogical, and geophysical data of the lunar surface. The mission has a scheduled lifetime of one year [2].

It is suggested that the Lunar Orbiter closely examine the proposed observatory site, Hertzsprung, and the alternate site, Mendeleev. It is hoped that further data will determine if Hertzsprung is indeed suitable for the FLARE astronomical observatory. This can be done by analyzing geographic data of the regions of interest. The main concern should be obtaining maps of high quality as well as accurate altitude maps. This data will allow for exact placement of the facilities that are now being designed for the far side lunar observatory.

2.3.2 Precursory Mission 2: Reconnaissance and Beacon Placement

Once the data gathered by the Lunar Orbiter has been analyzed and adequate maps have been assembled, three multi-purpose landers will be dispatched to the observatory site. The primary mission of the landers will be to place three radio beacons to assist the Lunar Operations Vehicle (LOV) carrying observatory materials to the lunar surface. The landers will be expected to have a maximum payload of 500 kg and the ability to land within 1 km of the chosen landing site without ground tracking. It is assumed these precursory mission landers will not be reusable. They will instead be disassembled and their parts will be used in the construction of the lunar observatory.

Each lander will carry a Radioisotope Thermoelectric Generator (RTG), a radio beacon, and a small suite of environmental and atmospheric instruments. They will be landed individually at the vertices of an equilateral triangle with sides of approximately 40 km in the northeast quadrant of Hertzsprung. The first lander will also carry a small 25-35 kg mini-rover similar to the one currently being tested by Jet Propulsion Laboratories (JPL) for the exploration of Mars. It will be equipped with a manipulator arm and a television camera and will be used to survey the site from the ground [6]. If the site is deemed undesirable after analyzing the data obtained by the rover and preliminary lander, further operations at the site will be terminated. If the site is confirmed to be favorable, as is expected, the second and third landers will land in turn at their designated locations.

With all three landers in place, the LOV can land safely at the exact locations by triangulating on the three radio beacons. The distance of 40 km between beacons is acceptable since the floor of Hertzsprung is anticipated to be bowl shaped [3]. Thus, the curvature of the lunar surface will not be a limiting factor for the lander/beacons system.

3.0 Discussion of Subsystems

Seven subsystems have been investigated in the design of the far side lunar observatory. They are as follows:

- Astronomical instruments
- Communications and power
- Habitation and transportation
- Cargo spacecraft design
- Trajectory analysis
- Thermal systems
- Robotics.

The following sections present an in depth analysis of each of these subsystems.

4.0 Astronomical Instruments

The focus of the FLARE base will be on its package of astronomical instruments. By means of these instruments, the FLARE project will be able to expand the horizons of the field of astronomy. The instruments package has consequently been designed to be not only a feasible part of a far side base, but also an effective surveyor of as many portions of the electromagnetic spectrum as possible.

4.1 Instrument Rankings

The establishment of a lunar far side base is quite a large undertaking. Roughly 170 metric tons of payload will have to be transferred from Low Earth Orbit (LEO) to the Moon's surface. Almost one-half of that mass will be instrumentation. Since funding or other constraints may dictate an overall reduction in the FLARE package at some future time, a ranking of all potential instruments has been made to determine the following:

1. Which instruments are essential to a far side lunar base?
2. Which are of lower priority to the project?
3. Which instruments should be dropped from the FLARE base scenarios?

In the decision matrix shown in Table 4.1, the primary criteria in ranking instruments were:

1. A measure of the instrument's scientific importance
2. A determination of how uniquely suited the instrument was to being placed on the lunar far side.

Consequently, the scores for these criteria were given greater mathematical weight in the final scoring. Therefore, an instrument that scored well in these two primary criteria usually overcame low scores in other criteria. For example, even though the optical interferometer is expected to be fairly massive and difficult to set up on the lunar surface, it scored highly because it offers a 10^4 improvement in resolution over the Hubble Space Telescope (HST). Furthermore, the use of the lunar surface as a stable base enables high position accuracy [1].

On the other hand, the neutrino detector and the infrared telescope scored below 20 in the ranking system. Hence, it is felt that they should be dropped from the package. The neutrino detector scored low primarily because it is best suited to being established at a more advanced lunar base than the L.S.A. has designed. The infrared telescope scored low primarily because it is better suited for emplacement within a crater near the lunar poles. Without perpetual natural shadow, it becomes more difficult to keep an infrared telescope at the proper temperature for operation. Also, the infrared telescope was the most massive of the "compact" astronomical instruments.

Using these types of rankings, seventeen instruments were chosen for inclusion in the FLARE project. Most of the selection is comprised of astronomical instruments, but some room has also been given to general survey packages, geological survey packages, and prototype materials processing facilities. The six astronomical instruments - all having ranking scores of more than 34 in the decision matrix - have been designated as being essential to the FLARE project. While other instruments may be dropped from the package, dropping these six instruments would severely undercut the entire purpose of the FLARE base.

Table 4.1. Instrument Package Decision Matrix.

Instruments	Criteria					Score
	Scien-tific (Wt: 1.25)	Unique to Farside (Wt: 1.5)	Mass (Wt: 1.0)	Operation Difficulty (Wt: .75)	Set-Up Difficulty (Wt: .75)	
Transit Telescope	8	9	9	10	7	45.25
MERI (15m radio tel.)	7	9	8	9	8	43.00
VLF Array	4	10	8	8	3	36.25
Optical Interferometer	10	8	6	4	2	34.5
Submillimeter Interferometer	9	8	5	6	2	34.25
16m Telescope	10	6	3	7	6	34.25
Environmental Survey Package	5	6	9	5	8	34.00
Magnetospheric Survey	4	5	10	5	10	33.75
Seismograph and Heat Flow Stations	5	7	7	6	7	33.50
Drive Shafts and Geophones	5	7	8	3	6	31.50
X-ray telescope	6	2	8	6	8	29.00
High Energy Cosmic ray Det.	7	5	6	4	4	28.25
Low Energy Cosmic Ray Det.	5	4	6	8	5	28.00
Mat.Plant I-Oxygen	4	6	7	2	3	24.75
Gamma-Ray Telescope	6	3	6	4	4	24.00
Mat. Plant II-HCl Leaching	3	5	6	2	3	21.00
Mat. Plant III-Carbonyls	3	5	6	2	3	21.00
Infrared Telescope	3	3	5	3	4	18.50
Neutrino Detector	5	1	1	6	1	14.00

4.2 Instrumentation Subsystem Overview

The instrumentation package will consist of 17 separate experiments, with a total mass of approximately 91.0 metric tons. Additionally, the instruments will demand 78.5 kWe of power and will require 52 kbps from the communications system of the FLARE base. Many of these packages will enable the FLARE project to survey multiple bands of the electromagnetic spectrum with heretofore unknown levels of accuracy. Other packages will enable small scale, but significant, studies to be made with regards to lunar geology and the feasibility of materials processing on the Moon. The parameters of the chosen instruments are summarized in Table 4.2.

Detailed descriptions of the structure, function, and requirements of each instrument are provided in the first three appendices to this report. Appendix A discusses each of the astronomical instruments, Appendix B describes the instruments to be used for geological survey work, and Appendix C presents the prototype materials processing plants that are to be put in place by FLARE.

Table 4.2. Instrument Package Parameters.

Instruments	Mass/ Volume	Power	Data Trans.	Cooling
VLF Array	2 metric ton	100 We	32 kbps	no
MERI (15m radio tel.)	2.1 metric tons	15,000 We	10,000 kbps	Yes,Pasive
Environmental Survey Package	100 kg	10 We	1 kbps	no
Magnetospheric Survey	10 kg	30 We	1kbps	no
Submillimeter Interferometer	14 metric tons	20,000 We	100 kbps	Yes, Passive, Active to 100° K
Transit Telescope	1.3 metric tons	400 We	30,000 kbps	Yes , Active 100° K
16m Telescope	42 metric tons	5000 We	10,000 kbps	Yes,Passive 100° K
Optical Interferometer	16 metric tons	9000 We	1000 kbps	Yes Passive/Active
High Energy Cosmic ray Det.	3 metric tons	1000 We	10 kbps	No
X-ray Telescope	2 metric tons	1000 We	100 kbps	No
Low Energy Cosmic Ray Det.	3 metric tons	500 We	10 kbps	No
Gamma-Ray Telescope	3 metric tons	1000 We	25 kbps	No
Mat.Plant I-Oxygen	.565 metric ton	12,900 We	unknown	No
Mat. Plant II-HCl Leaching	.695 metric tons	6,500 We	unknown	No
Mat. Plant III-Carbonyls	.5 metric tons	6,000 We	unknown	No
Seismograph and Heat Flow Stations	.5 metric tons		unknown	No
Coring Drive Shafts and Geophones	.25 metric tons		unknown	No

Subtotals	109.92 metric tons	78.55 kWe	51,280 kbps

5.0 Communications

The communication system is an integral part of the far side lunar observatory. There are several requirements which must be met in order to properly operate a lunar base. They are as follows:

- Constant communication from the lunar base to Earth is needed for telecommunication and telerobotics.
- The communication system must be able to transmit large amounts of data and information.
- The communication system must offer several video channels for use with telerobotics.

The communication system provides the essential data, audio, and video transmissions to construct and operate a lunar observatory. This communication will allow for scientific experiments, lunar construction, lunar observation, and a manned presence. The system will include a lunar surface terminal which will serve as a center for the receiving, transmission, and compression of information.

The communication system that was considered has an estimated transmission rate of 50 Mbps. In order to meet the needs of the future, a data relay satellite system must be utilized to meet the requirements of high data rates. Furthermore, in order to send a large amount of information over a signal, the signal must be transmitted at a high frequency. Current communication satellites do not meet these strict requirements. However, a new satellite system, ATDRSS (Advanced Tracking and Data Relay Satellite System), is currently under development by NASA Goddard Research Center and does meet the requirements.

The ATDRSS boasts three tri-band single access antennae which allow three users to simultaneously use each antenna. The ATDRSS can operate over the Ka band in the 20 - 30 GHz range. The forward link operates between 22.55 - 23.55 GHz and has a data rate of 50 Mbps. The return link operates between the 25.25 - 27.50 GHz range and has a data rate of 650 Mbps [1]. Furthermore, the ATDRSS has the capability to communicate from an Earth based station to a space-based station, whereas current communication satellites can only receive and/or transmit to Earth stations.

This ability of the satellite to "look both up and down" is beneficial for the lunar mission.

Additionally, it is necessary for the communication satellite to have an optical lens with a resolution of 1 m for use in tracking robotic instruments. It is assumed that the ATDRSS satellite, or a similar substitute, will have this type of optical lens as well as antennas which can produce 60 db of gain. The satellites must also carry transponders with a maximum power output of 1 kWe.

The requirement for constant communication with a far side lunar base during construction and operations is critical. The communication system must be able to transmit and receive video images, audio signals, and digital signals both to and from Earth. However, since the base is located on the far side of the Moon, a direct line of sight to Earth is not possible. Several satellite transmitter and receiver locations were studied to provide the most economical means of achieving constant communication with Earth. The communications needs were then analyzed as two separate problems - communications during the construction phase and communication during the operational phase.

The first phase of the mission is the construction phase. During this phase, the immediate establishment of a communication system to allow for construction of the base is the main objective. The second phase is the operational phase. It is during this phase that the base is fully functional and the astronomical studies are ready to proceed. It is important during this phase that the communication system does not interfere with instrument readings.

5.1 Construction Phase

Three location scenarios for the relay satellite were studied for communication during the construction phase. These scenarios included placing a data relay satellite (DRS) at the L4 or L5 libration point, placing a DRS in a halo orbit around the L2 libration point, or placing three DRSs in polar orbit about the Moon. From the data relay satellite, the signal will be directed toward a geostationary communications satellite, similar to the ATDRSS, which can efficiently transmit signals to Earth.

5.1.1 Use of the L4 Point and Relay Satellite

The L4 libration point was selected as a possible location for a relay satellite for several reasons. Since the L4 libration point is fairly stable, only temporary spacecraft attitude adjustments are necessary. A satellite in the L4 point would also provide a direct line of sight with the Earth. However, there is not a direct line of sight from the Hertzsprung base location to the L4 point. Therefore, this type of arrangement would require an antenna to be placed near the limb of the Moon in order to achieve a line of sight with this point. Furthermore, any base locations not near the limb would require a relay antenna system along the lunar surface in order to broadcast to the L4 point. During the construction phase, establishing this type of relay communication system would not be a feasible communications option.

5.1.2 Satellites in Low Lunar Orbit

Another communications option that was studied was placing three communication satellites into Low Lunar Orbit (LLO). These three orbiting satellites would always be in position relative to one another such that they would be able to broadcast directly to Earth from any lunar location. The major disadvantage to this system is the requirement of placing three satellites into orbit rather than just one. Additionally, the lunar tracking station would have to reacquire the active communication satellite periodically, temporarily breaking continuous communications with Earth.

5.1.3 Halo Orbit

Still another communication option that was studied was placing the satellite in a halo orbit about the L2 libration point. The L2 point lies on the Earth-Moon line, approximately 64,500 km beyond the Moon [2]. However, the L2 libration point is unstable and would require translational control to keep the satellite in the vicinity. Furthermore, in order to avoid the occulted zone caused by the Moon, the halo orbit must have a radius of 4000 km [3]. A halo orbit of this size would require a ΔV (change in velocity) of 426 fps/yr [3]. However, the halo orbit would allow constant communication with the Earth and could be reached by most locations on the far side of the Moon. Because the halo orbit option allows continuous communication with the Earth using only one relay satellite, it has been chosen as the primary communication option during the construction phase of the mission. Figure 5.1 shows the geometry of the halo orbit.

5.1.4 Construction Phase Communication Decision Matrix

A decision matrix was generated to help determine the best communications system to be used during the construction phase. Table 5.1 shows this decision matrix. From this figure, it is clear that the best choice for communications during the construction phase is the use of a satellite in an L2 halo orbit.

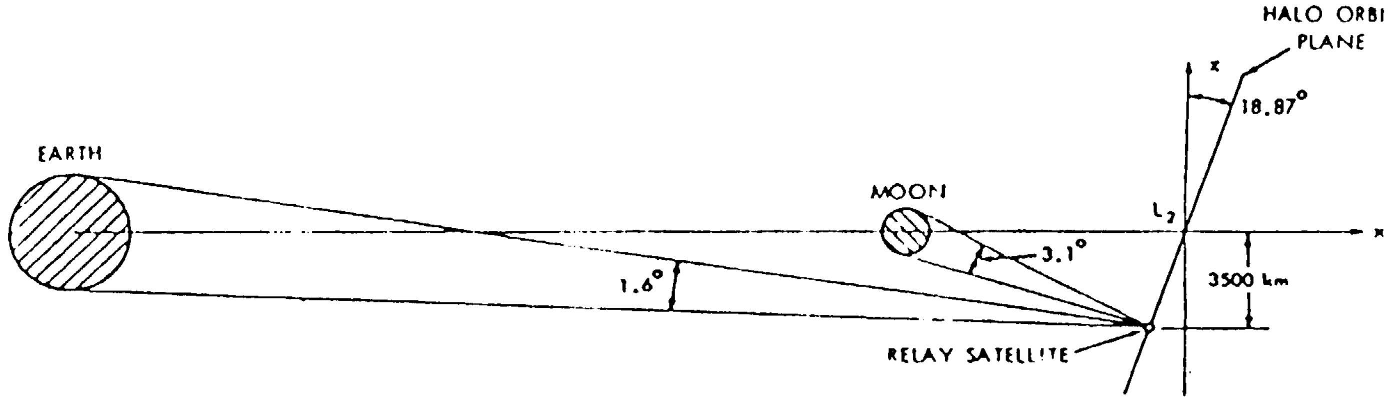
Table 5.1. Decision Matrix for Construction Phase Communication.

Alternatives:	S/C at L4	S/C in L2 Halo	3 S/C in LLO
Hardware Required	8	8	2
Initiation Ease	1	8	6
Maintainability	9	6	3
Total	**18**	**22**	**11**

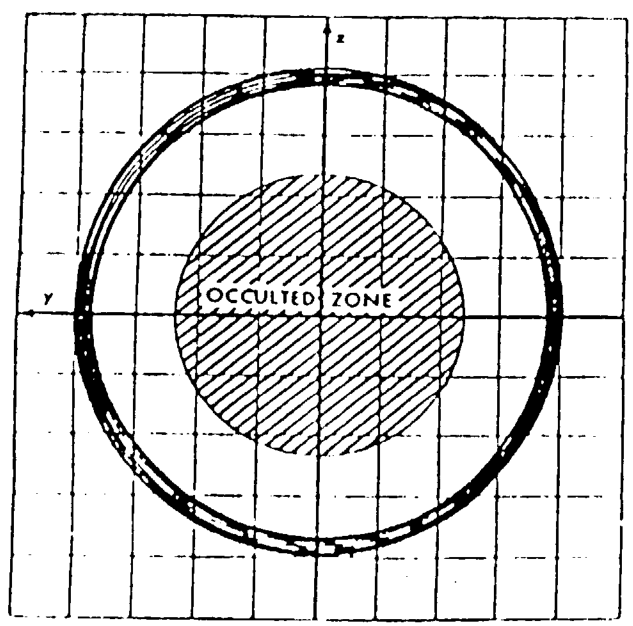
5.2 Operational Phase

Once the base is operational and the instruments are ready to begin testing, communications will have to be handled in a different manner. A satellite broadcasting in a halo orbit above the base would interfere with some of the radio frequencies that are trying to be observed. In order to have a radio-free sky for astronomical observations, it would be desirable to have a communication system that would be able to broadcast and not interfere the astronomical readings.

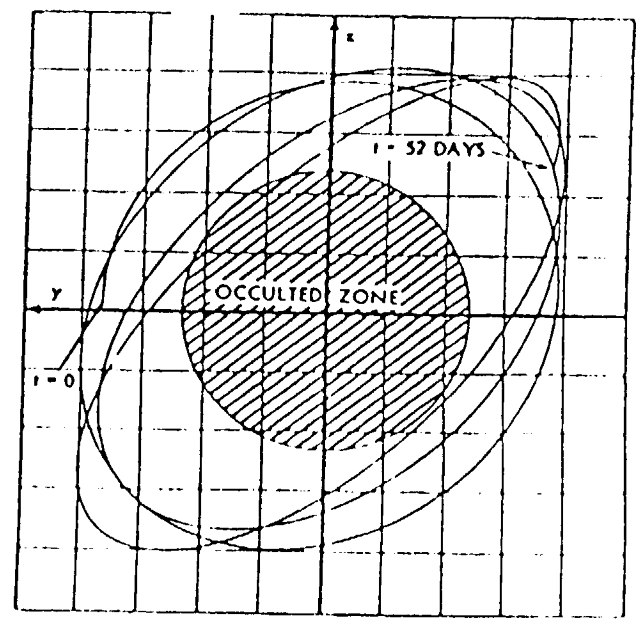
Establishing a communication station to broadcast from a point beyond the limb of the Moon directly to the Earth would alleviate this undesirable situation. From the chosen base location in Hertzsprung, the edge of the Moon is approximately 1030 km away. In order to compensate for lunar wobbling, a communication station should be placed at least 1200 km away from the base on the limb. Difficulties arise in designing an operational communication system that would transmit a



Halo Orbit Configuration



Halo Orbit With Stationkeeping



Halo Orbit Without Stationkeeping

Figure 5.1. Halo Orbit.

signal which must be relayed over a distance of 1200 km. The use of either microwave relay towers or fiber optic cables were options that were studied as the methods of relaying the data from the lunar base to the communication station on the limb of the Moon.

5.2.1 Microwave Relay Towers

Microwave relay towers are a very effective and widely used means of signal relay on the Earth. However, since the curvature of the Moon is greater than that of the Earth, transferral of a signal would require a large number of towers over a relatively short distance. From Table 5.2, it can be seen that a system of towers 200 feet tall would require 26 microwave relay towers placed every 47 km apart to span a distance of 1200 km.

Table 5.2. Microwave Relay Towers.

Tower height	Distance between towers (km)	Number of towers required	Tower mass (kg)	Sum of tower mass (kg)	Sum of dish mass (kg)	Total mass (kg)
100	32	38	288	10944	2585	13529
150	40.2	30	391	11730	2100	13830
200	47.34	26	514	13364	1820	15184

5.2.2 Fiber Optic Cable

The use of fiber optic cable for communications is a relatively new tool in the communications world and has not been tested to a large extent in outer space. The type of cable that would seem most appropriate for an outer space mission is tactical fiber optic cable.

The tactical cable offers many advantages over common cables used today. The cable is radiation hardened which prevents the fibers from experiencing any outgassing effects due to the lack of atmosphere. A new process of doping the fiber with Euribidium allows the signal to travel further before regeneration is needed. Current tactical fiber optic cable requires the signal to be regenerated every 60 miles (96 km). Each fiber has a data rate capacity of 565 Mbps [4]. Another advantage is that fiber optic cable shows no adverse effects when subjected to electro-magnetic radiation. AT&T has estimated an indefinite lifetime for fiber optic cable and the potential of its future uses seems limitless [4].

Difficulties might arise when laying the cable across the rugged terrain of the Moon. The Mobile Surface Application Transverse Vehicle (MOSAP - see § 6.7) can travel a distance of 3000 km and could possibly spool out the necessary cable as the rover travels this distance. One precaution that must be taken is avoiding any sharp bends or kinks in the cable as this will restrict the data flow similar to the way in which a kink restricts the flow of water from a garden hose.

The future of fiber optics is becoming even more enticing as a possible communication system. The data rate capacity of fibers in the near future is estimated at 1.7 Gbps [5]. Fiber optic cable has demonstrated its ability to withstand large temperature differences without much signal degradation.

5.2.3 Choice of Operational Communications System

From the data gathered on both the microwave tower and fiber optics, an operational communications system was chosen. Table 5.3 shows the decision matrix used in choosing the operational communication system. From this table, it is clear that the best choice of communication system during the operational phase of the lunar base would be a system using fiber optic cable.

Table 5.3 Operational Phase Communication Decision Matrix.

Alternatives:	Microwave Relay Towers	Fiber Optic Cable
Construction Ease	4	9
Payload Size	7	7
Number of Relays Needed for 1200 km	5	7
Total	**16**	**23**

5.3 Moon Station

A few assumptions had to be made when determining the link budget for the communications system. For the lunar communication station shown in Figure 5.2, a 25 m parabolic dish was selected in order to meet the high gain requirements needed for the system. This communication system also requires a great deal of power to boost the signal gain to a high enough level that it can be heard over the noise and space loss. The majority of the signal decrease is due to the space loss. Space loss is the decrease in power of the signal due to the large distances the signal must travel. The requirements of the communication system are shown in the Table 5.4. The moon station receiving quality ,G/T, has a value of 2.96 db/K and the Carrier/Noise ratio is 78.56 db-Hz.

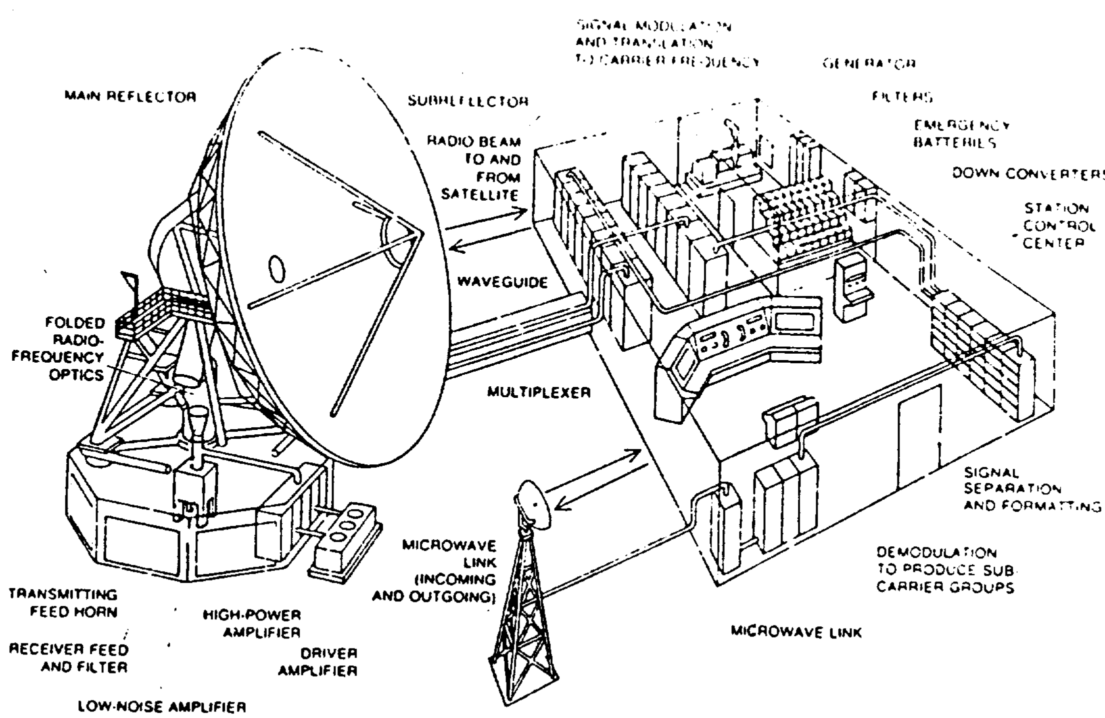


Figure 5.2 Communications Terminal on the Lunar Surface [2].

Communication Station Link Budget

Uplink

Frequency	23	Ghz
Ground EIRP	135	db
Space Loss	-210	db
Misc. Losses	-2	db

Downlink

Frequency	27	Ghz
S/C EIRP	122	db
Space Loss	-232	db
Misc. Losses	-1	db
Ground G/T	2.96	db/K
Channel Noise	-54	db
Downlink C/N	78.56	db-Hz

5.3.1 Signal Path

The power for the signal path is shown is the Figure 5.3. The signal starts at a level of -30 db which reduces the overall power requirements of the transponders. As mentioned before, the space loss accounts for most of the decrease in the signal power. Miscellaneous losses such as line feed, system degradation, and rain attenuation have been neglected in this figure as they are negligible in comparison to the space losses.

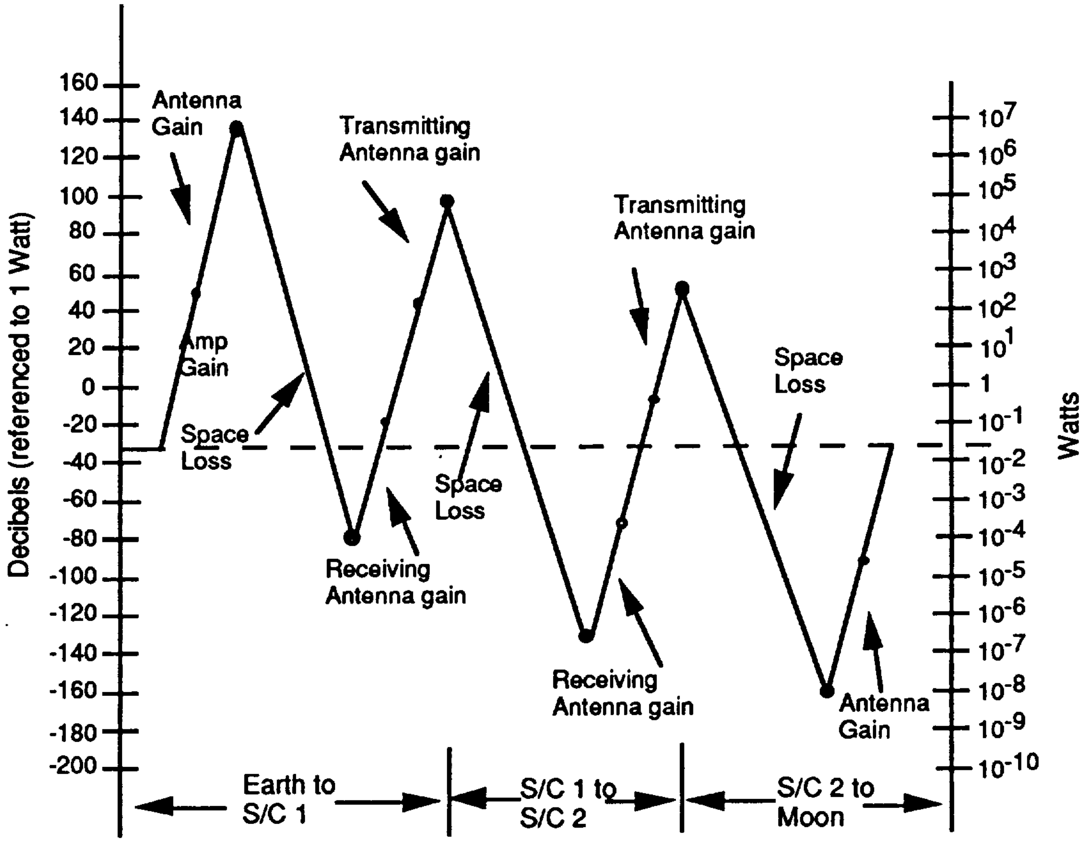


Figure 5.3. Signal Power Budget.

Further information on noise and data compression can be found in Appendix D.

5.4 Communication System Overview

To summarize the communication subsystem, a satellite placed in an L2 halo orbit will be used for communications while the lunar base is under construction. It will offer constant communication with Earth and will be accessible from all locations of the far side of the Moon. During this stage, the lunar station terminal will consist of a 25 m parabolic dish and require an output power of 1 kWe.

During the operational phase, however, a new broadcast location will have to be used since broadcasting to the L2 halo orbit satellite would not provide a radio-clear sky. Therefore, during the operational phase of the base, a fiber optic cable will be used for communication. It will span a distance of nearly 1200 km to this new broadcast location on the limb of the Moon.

5.5 Power System

During the construction phase of the mission, successful establishment of a power system is crucial. To meet the initial needs of the lunar base, solar arrays will be used as a temporary power source with each instrument package having an autonomous power source until a central power system can be established. It is estimated that a two to three year construction period will be required until the permanent power system achieves operational status. Additionally, the robotics needed to set-up the base will be powered by fuel-cells and recharged by the temporary solar arrays.

The total power requirement for the base has been estimated to be approximately 160 kWe. Figure 5.4 illustrates the power systems available for a given power need. This chart clearly shows that if the power requirements of the base are greater than 150 kWe, then the only viable power option is nuclear power. If the actual power requirements were on the order of 100 kWe, then either solar power or nuclear power could serve as a feasible alternative.

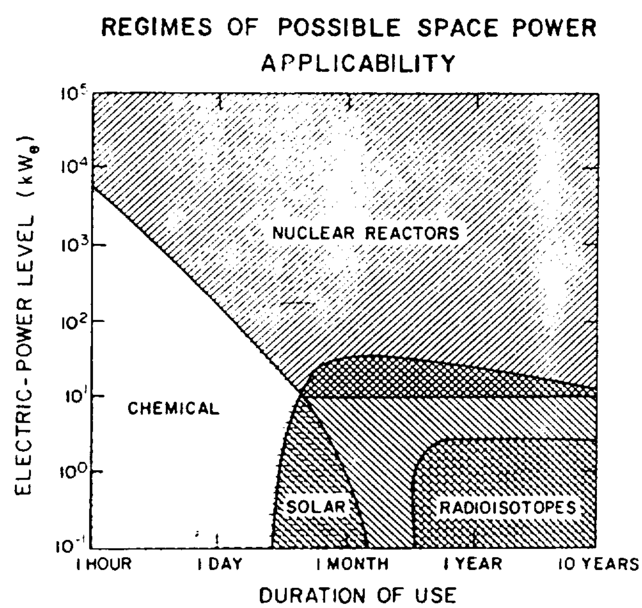


Figure 5.4. Applicability of Power-Technology Options to Space Mission of Various Power Requirements and Orbital Duration [1].

5.5.1 Solar Arrays

Solar arrays are able to produce 250 We/m^2 with a power density of 28 We/kg, but they have a yearly power depreciation of 4% per year and an expected lifetime of 7 years [2]. The use of solar arrays in space is quite extensive and therefore the technology is well known. The problem in dealing with solar power with respect to the FLARE base is meeting the large power requirements of the base during the lunar night. Since the base is located at the equator, the panels will experience 14 Earth days of light and 14 Earth days of darkness. In order to continue supplying power to the base during the lunar night, fuel cells charged during the lunar day would have to be utilized. The solar arrays would, therefore, have to produce twice the power that is actually required by the base so that the fuel cells could be charged sufficiently while still maintaining base power. This fact would also mean that twice as many panels would have to be used than would normally be required. For the thermal control of photovoltaic arrays, simple radiators must be used due to the non-convective nature of the space environment.

5.5.2 Nuclear Power

Nuclear power is an attractive means of supplying electricity in that nuclear plants are able to generate large amounts of electricity at a higher specific power output than photovoltaics can manage. Two types of nuclear reactors were studied as a possible sources of permanent power for the far side lunar observatory. They are the Soviet built Topaz and the SP-100, a joint project of the Department of Energy (DOE), NASA, and the Defense Advanced Research Projects Agency (DARPA). The Topaz has a power output of 10 - 15 kWe and a specific power of 15 We/kg [3] with a unit mass of one metric ton. This implies that the FLARE base would require 16 units. The SP-100 has a designed power output of 100 kWe and a specific power output of 33 We/kg with a unit mass of five metric tons. If this option were utilized, two units would be required. The SP-100 has a life expectancy of 10 years [3].

One problem in dealing with nuclear power is the complex thermal controls needed to maintain the plant. Radiator design depends on operating temperature, working fluid selection, and power level. The amount of waste heat that can be radiated to space by a given surface area is determined by the Stefan-Boltzmann law which states that the waste heat that can be radiated by a blackbody surface is proportional to the fourth power of the radiating surface's temperature. The surface and mass of the thermal radiator are very sensitive to the heat rejection temperature. Higher heat rejection temperatures correspond to smaller radiator areas and thus lower radiator masses [1].

Another problem with nuclear reactors arises in trying to deal with radiation. The use of a nuclear power source in space is frequently accompanied by the need for a radiation shield to protect the astronaut crew and payload from the detrimental effects of radiation. This is especially true for nuclear reactor systems where copious quantities of neutrons and gamma rays accompany the fission process. The establishment of permissible radiation levels to protect personnel and/or sensitive equipment is the major focus in the design of a radiation shield for a base location.

Many factors influence the structure, composition, and mass of the radiation shield. Some of these factors include:

- The size and nature of the nuclear power source
- The type of radiation - alpha, beta, gamma, neutron, etc.
- The configuration of the spacecraft or platform and its payload (including crew compartment)
- The mission-generic operational procedures and requirements - such as the need for rendezvous and docking or on-orbit maintenance
- The length of the mission
- The total radiation dose levels that will be permitted.

5.6 Choice of Power System

The decision matrix shown in Table 5.4 shows how the operational power system was chosen. After analyzing the factors shown in this figure and weighing each option's advantages and disadvantages, it was decided that four clusters of four Topaz reactors would best suit the needs of the base. In case of a shut down of one of the Topaz reactors, fifteen will still be functional and a net power loss of only 10 kWe would be experienced. Since the reactors will be arranged in clusters, a power loss of only 40 kWe would be experienced in the event of a meltdown of one of the clusters. The 120 kWe of power remaining would be enough to sustain minimal operations on the base until assistance could be provided. Additionally, each cluster will be thermally independent.

Table 5.4. Power System Decision Matrix.

Category	Topaz	SP-100	Solar
Specific Power	6	8	4
Volume	6	9	5
Feasibility	10	10	2
Power Degradation	9	9	3
Durability	10	9	5
Contingency Problems	9	1	10
Set-Up Difficulties	4	5	8
Cost	6	4	8
Total	**60**	**55**	**45**

5.7 Reactor Thermal Control

The thermal control for a space nuclear reactor consists of two parts, the primary heat transport system which cools the reactor core, and the waste heat rejection system which eliminates heat that has not been converted to electricity or utilized for base purposes. The following sections detail the analysis of these two systems.

5.7.1 Primary Heat Transport

The primary heat transport system in a space nuclear power plant carries thermal energy from the nuclear heat source to the point of application. Excess thermal energy that is not converted into electricity must be rejected to space by means of a thermal radiator. The heat transport system can consist of either one of two general types: pumped-loop or heat pipe.

Pumped-Loop Heat Transport System:

Pumped-loop heat transport systems include the all-liquid working fluid loop, the liquid-vapor working fluid loop, and the gaseous working fluid loop. In the primary reactor loop, an all-liquid working fluid loop is simply a heat transport loop, while liquid-vapor and certain gaseous working fluid loops may contain the power conversion subsystem working fluid. Heat pipes achieve the transport of thermal energy by the evaporation, condensation, and return capillary action of a two-phase working fluid in a sealed container. The heat pipe is self-contained and requires no pumps, compressors, or external power to achieve large quantities of nearly isothermal heat transport.

Furthermore, the fluids in pumped-loop heat transport subsystems are generally metals in either the liquid or vapor phase or gaseous form. These working fluids are typically transported in thin-walled, metal alloy tubing. For a typical primary heat transport loop, the working fluid is circulated from the nuclear heat source (the reactor) to either a heat exchanger or the power conversion subsystem and then returned to the heat source. For nuclear heat sources (especially reactors) with high thermal fluxes and high operating temperatures, liquid metals are of special interest as working fluids. These metals have superior thermal conductivity and low vapor pressure. In addition, they are stable at high temperatures and in intense radiation fields. Low atomic weight metals such as lithium and sodium also have relatively high specific heats and volumetric capacities.

A liquid metal loop's piping must have a method of accommodating the liquid metal inventory, thermal expansion, and contraction of the loop itself. This is necessary to maintain a void-free system at appropriate operating pressures. Volume accumulators were developed for this purpose in the space nuclear power program. The accumulator consists of a bellows assembly that is designed to keep a volume of liquid metal in compression by means of a either a mechanical or gas spring. The accumulator is generally located in a relatively low temperature region of the loop and connected to the loop by means of small diameter tubing.

Heat Pipe Heat Transport System:

The heat pipe heat transport system is a self-contained device that is used where high thermal energy transfer rates are needed or where precise temperature control of a process is required. The heat pipe attains very high thermal conductances by means of two-phase fluid flow with capillary circulation. In ordinary conduction heat transfer, thermal energy is transported by the motion of atoms and electrons. Metals, due to their special atomic structure, have exceptionally high thermal conductivities. Silver, for example, has the highest known thermal conductivity of all metals, yet the effective thermal conductivity of a heat pipe can be thousands of times greater than that of solid silver.

The high thermal conductance property of the heat pipe is achieved by making use of the latent heat of vaporization of the working fluid. As shown in Figure 5.5, an operating heat pipe is basically a container in which the working fluid circulates between heated and cooled regions. In the heat pipe, this fluid exists as two phases: liquid and vapor. In the heated or evaporator region, thermal energy transfer to the liquid causes it to experience phase change and become a vapor. The vaporized working fluid then flows from the evaporator region to the heat pipe's cooled, or condenser, region. There, the thermal energy transfer process is reversed. As the vapor cools, it becomes a liquid and loses energy. A wick structure within the heat pipe provides capillary action, permitting the return of the liquid to the evaporator region. The ability of a heat pipe to transfer thermal energy is proportional to the latent heat of vaporization of the working fluid. Typical values range from 190 kJ/kg for freon to 4370 kJ/kg for sodium. However, lithium also has an extremely high latent heat of vaporization, 20,525 kJ/kg.

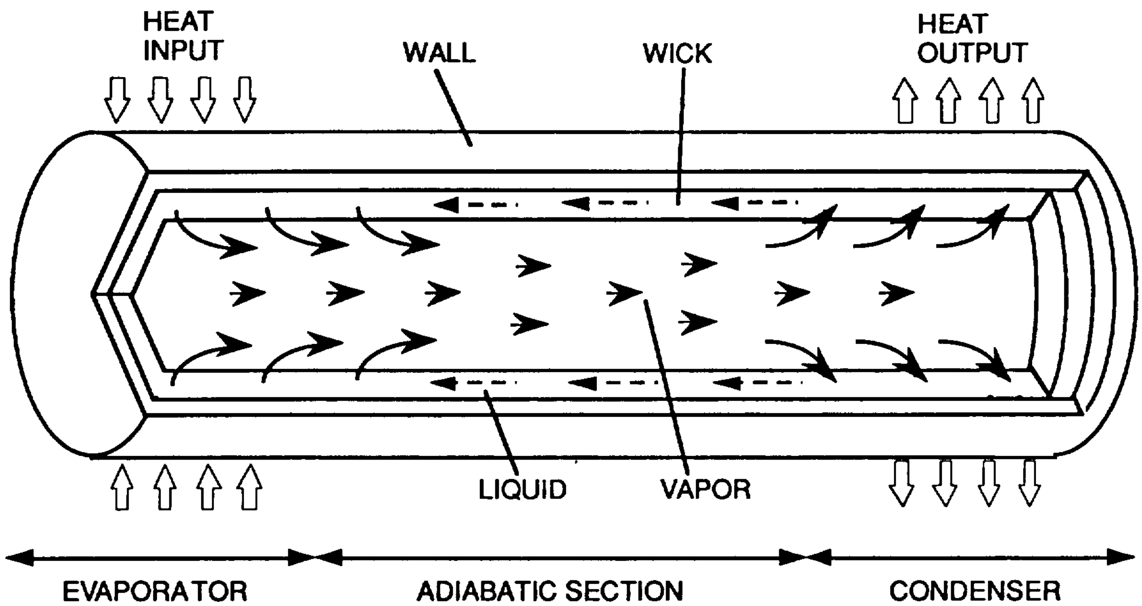


Figure 5.5. Heat Pipe Operating Principle [1].

The high temperature heat pipe has a number of technical features that makes it especially attractive as a key component in advanced space nuclear reactor systems. First, and perhaps foremost, such heat pipes represent a very high thermal energy transport capability in a relatively simple, self-contained device capable of high temperature operation. Reactor cores may thus be designed with many independent cooling loops, the failure of any one of which would not jeopardize the safe operation of the space nuclear power plant. The elimination of such single-point failure mechanisms in the primary reactor coolant system establishes a power plant design environment essentially devoid of catastrophic failure modes. This is an extremely important technical feature for space power systems that may be required to operate reliably for five or ten years without failure [1].

The FLARE mission will utilize a lithium heat pump for the primary heat rejection of the nuclear base. The pipes will be approximately two cm in diameter, two meters long per pipe, with an temperature of 1270 K, and about 500 kg total mass per cluster.

5.7.2 Waste Heat Rejection

The waste heat generated by the space nuclear power plant must be rejected by radiation due to the non-convective nature of space. Radiator design depends on both the operating temperature and the power level of the reactor. Additionally, the surface area and mass of the thermal radiator are very sensitive to the heat rejection temperature. Figure 5.6 shows the relationship of mass and area as a function of the radiator temperature.

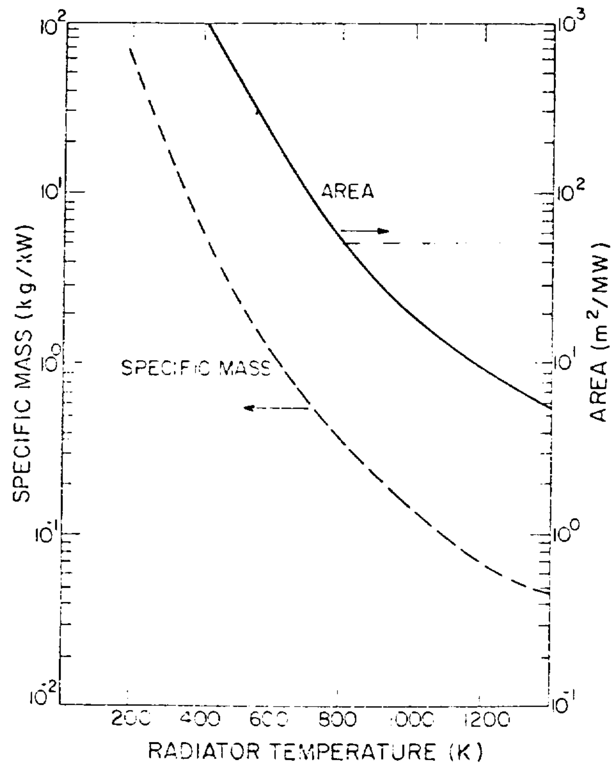


Figure 5.6. Radiator Area and Mass as a Function of Temperature [1].

The heat pipe can be used in a variety of configurations in the thermal radiator design. Figure 5.7 shows a heat pipe-fin scheme in which thermal energy to be rejected is received at the evaporator region of the pipe, transported isothermally to the condenser region, and then radiated away to space after conduction through the wall to the flat fin. The heat pipe can either have high emissivity coating or else be attached to a high-emissivity fin that serves as the radiator surface.

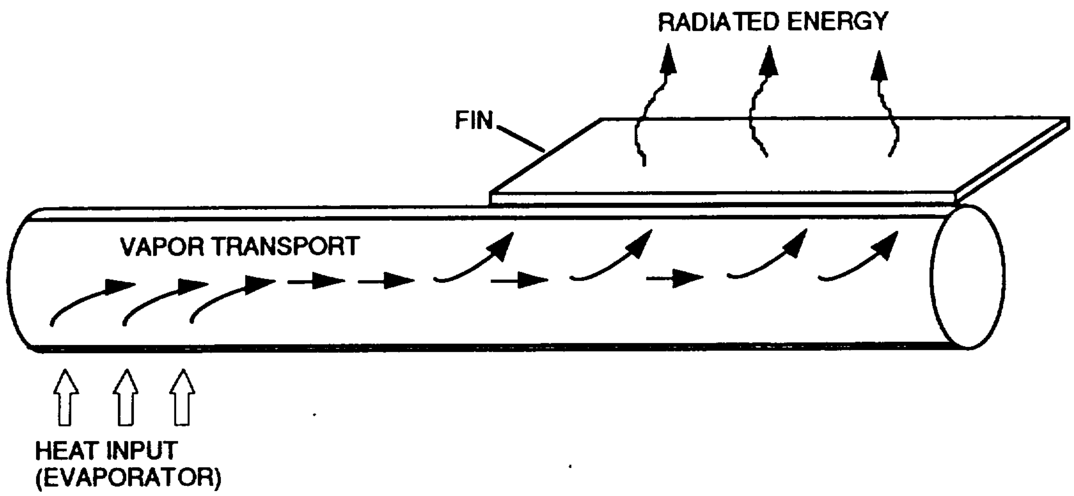


Figure 5.7. Heat Pipe-Fin Radiator Configuration [1].

A generic liquid metal pumped-loop heat rejection arrangement is shown in Figure 5.8. This configuration includes the following components:

1. Waste Heat Exchanger
2. Radiating Surface
3. Volume Accumulator
4. Electromagnetic (EM) Pump.

The volume accumulator serves two functions in such a closed loop. First, it compensates for thermal expansion of both the fluid and container from startup to operating temperature. Second, it keeps the fluid at a preselected pressure. Unlike heat pipe radiators, the pumped-loop heat rejection systems suffer temperature losses around the loop. This non-isothermal condition forces the radiator to operate below the waste heat rejection temperature of the power conversion system with a subsequent penalty of a larger radiator area.

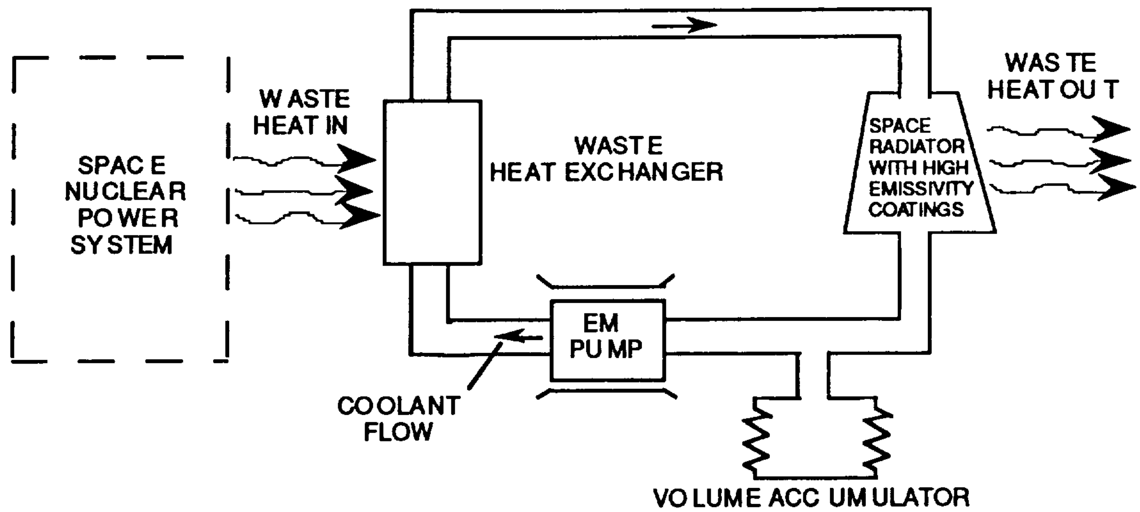


Figure 5.8. Generic Liquid Metal Pumped-Loop Radiator Configuration [1].

Due to the simplicity of design and fewer working parts, the heat pipe-fin combination will be used for waste heat rejection. The radiators will have an operating temperature of 780 K and will be constructed of titanium due to its high strength-to-mass ratio, high resistance to meteoroid damage, compatibility with the working fluid, and good structural integrity [1].

5.8 Power System Overview

During the construction phase of the lunar base (the first two to three years of construction), solar arrays will provide power to essential systems. Once the base is operational, however, four clusters with four Topaz reactors in each cluster will be used to power the base. These reactors will supply 160 kWe to the base and have desirable emergency/contingency plan characteristics. Heat pipes will be used for primary core cooling and a heat pipe-fin combination will be used for waste heat rejection.

Given the current political relationship that the United States holds with the Soviet Union, the present economic state of the U.S.S.R., and the clearance already given to purchase one reactor for scientific purposes, it is predicted that the United States will have no trouble purchasing the Topaz reactors from the Soviet government.

6.0 Habitation and Lunar Transportation

The habitation system includes the modules in which the crew members live as well as the airlocks used for ingress and egress. Other considerations taken into account include radiation shielding, meteorite shielding, and dust control of the habitat. The lunar transportation requirements for the crew are also presented.

6.1 Habitat Options

Lone Star Aerospace considered two modular options for the habitation system. These two habitation options were considered due to their simplicity as they are both prefabricated and modular in structure. These modules would allow for reduced construction time and cost, as well as easy implementation and expansion. The two options that were reviewed are the inflatable Module (IM) and a derivative of the space station common module (SSCM).

6.1.1 Inflatable Module

The inflatable module which L.S.A. considered for the habitat was designed by Eagle Engineering [1]. This habitat has five levels: four where the crew can live and work and a fifth level for stowage and the environmental control and life support system (ECLSS). The inflatable habitat is made up of two structural systems. The primary structural system supports the loads from a pressurization of the habitat to 14.7 psia and consists of the spherical envelope that surrounds the habitat. The secondary structure supports the loads from the crew, furnishings, and equipment. More information about the inflatable module is provided in Appendix E.

6.1.2 Space Station Common Module

The other option investigated by L.S.A. for the habitat module is a derivative of the Space Station Common Module (SSCM). Lone Star Aerospace decided to use two SSCMs for the lunar base after analyzing the advantages and disadvantages of both options. It was decided to use two SSCMs since using one module would cramp the inhabitants and be uncomfortable for the crew [2]. These two SSCMs will be connected end to end by a node.

The SSCM was designed so that it could be transported by the shuttle. As a result, its design was based on the assumption that the module would fit inside the payload bay. The original cylindrical pressure-vessel design - in terms of structural layout, materials selection, and nodal configuration - will be preserved for the lunar application, although modifications of crew accommodations will be necessary for SSCM adaptation to the lunar surface.

The SSCMs are cylindrically shaped and have a length of 13.57 meters. The approximate gross payload weight for the SSCM is 19,500 kg. The geometry and overall dimensions of the SSCM are shown in Figure 6.1 [2].

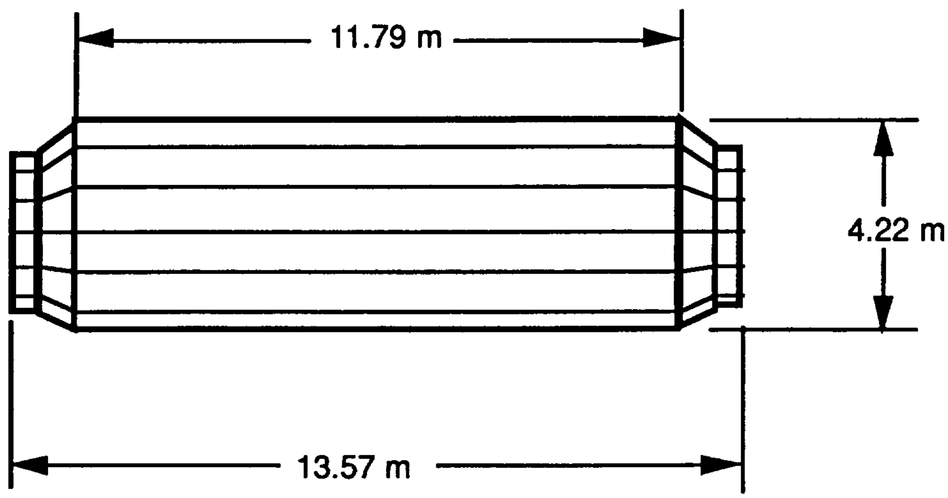


Figure 6.1. Space Station Common Module Specifications.

One of the advantages of the SSCM is that it is easy to simulate an Earth-like environment with respect to the room configuration. A cross-section of this configuration can be described geometrically as a rectangle inscribed in a circle. The rectangular area provides the illusion of "floors" and "ceilings " to the crew. Figure 6.2 presents an end cutaway view of a generic module.

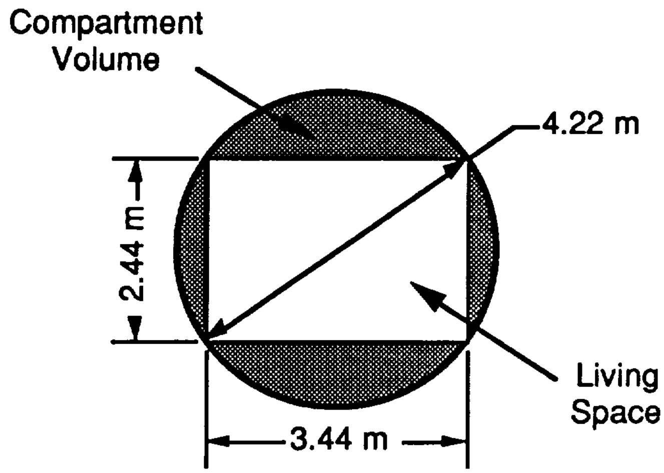


Figure 6.2. Cross-Sectional Area of the SSCM.

To accommodate added spring in the crew's step, a ceiling height of 2.44 meters has been selected. As a result, the interior space of the habitation facility is transformed into a rectangular volume, and is similar to terrestrial standards for the crew. This total compartment volume for the effective cylinder length of 11.79 meters is 58.35 cubic meters. The semicircular compartments around the sides of the rectangular area are designed for stowage, to house the ECLSS equipment, the electrical wiring, the plumbing, and the thermal control loops, as well as food and other supplies. Figure 6.3 provides a sample layout of the module interior from an overhead perspective [3].

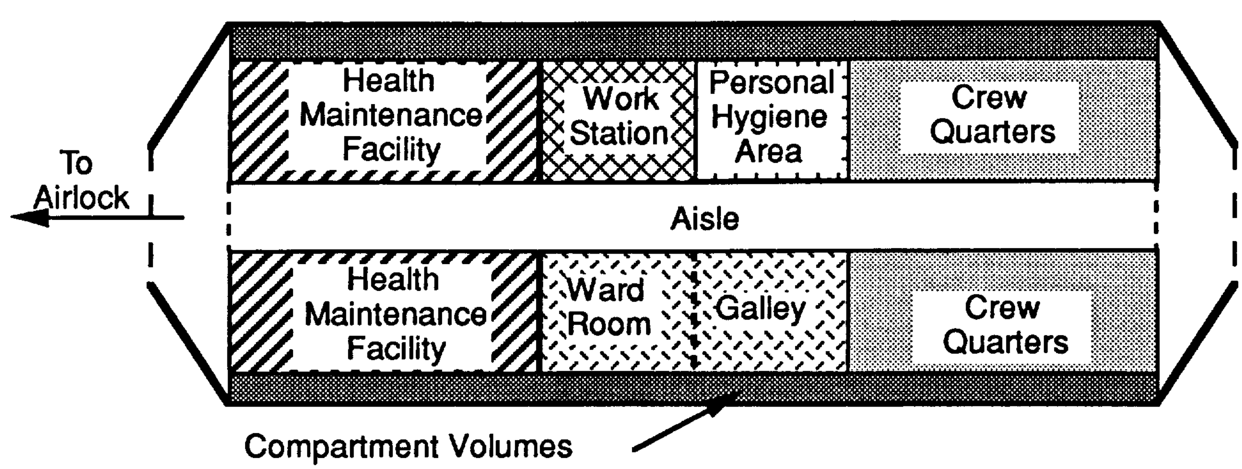


Figure 6.3. SSCM Interior Layout.

Configuring a module for the basic living requirements of the lunar base crew entails redesign of the SSCM dimensions. Extending the length of the module, rather than the diameter, would not involve serious manufacturing changes. In fact, the lunar base could conceivably consist of a variety of different length modules, depending on the intent of the modules and the number of crew members. Table 6.1 details the SSCM interior specifications [3].

Table 6.1. Adjusted Rectangular Dimensionsof the SSCM.

Height (floor-ceiling), m	2.44
Width, m	3.44
Length, m	11.79
Adjusted Interior Volume, m^3	98.82
Side/Wall Compartment Volume, m^3	11.62
Floor/Ceiling Compartment Volume, m^3	46.72
Total Compartment Volume, m^3	58.35

6.2 Choice of Habitation Module

A decision matrix was used to determine the best habitation option for the lunar base. Using the criteria shown in Table 6.2, the SSCM was found to be the better habitation option for the use in conjunction with the FLARE base.

Table 6.2. Habitation Module Decision Matrix.

Criteria	Weighing Factor	IM	SSCM
Structural Integrity	10	1	2
Mass	15	1	2
Expandability	15	1	2
Living Space	10	2	1
Storage Space	10	2	1
Assembly	15	1	2
Design Flexibility	10	1	2
Transportation	15	1	2
Total	**100**	**120**	**170**

6.3 Ingress and Egress from the Habitation Module

Two minimum loss airlocks were chosen to provide the crew access to the lunar surface. These airlocks will be one-man airlocks each consisting of a vertical entry and a two-door cylindrical airlock with a staging chamber. The user will enter and exit in a vertical position. A circular escape hatch on top of the airlock will provide an exit in case of malfunction. The doors will be manually operated and will be designed to be easily be opened and closed by one person. If an emergency arises, the staging chamber can also be used as an airlock

The airlock will use an O-ring seal placed within a machined dove-tail groove to provide an air tight seal. This O-ring seal will be made of Kel-f plastic, a teflon polymer A rotation seal will be used for sealing the door locking system.

The status of the airlock door will be determined by four magnetic proximity sensors which will be continuously monitored by the habitat computer. Furthermore, the temperature and pressure inside the airlock will be measured by thermocouples and strain gage pressure sensors, respectively. The thermocouples and strain gage pressure sensors will also be continuously monitored by the habitat computer. The computer will sample the pressure inside the airlock at predetermined intervals. If the computer senses a dropping trend in the airlock pressure, it will warn the crew of air loss. A sketch of the airlock is shown in Figure 6.4.

6.4 Dust Control

To prevent any dust from entering the airlock (and subsequently the habitation module), a "mud room" will be placed outside the airlock. This room will be equipped with alpha particle emitters to neutralize any static charge build up on garments that would attract dust particles. Furthermore, crew members returning from outside the habitat will also be required to use a dust brush, similar to those used in Apollo missions, to remove any dust not removed by the alpha-particle emitters prior to entering the airlock.

6.5 Radiation Shielding

One of the most dangerous aspects of any manned exploration is exposure to radiation. The crew of the far side lunar observatory will be exposed to two types of radiation: Galactic Cosmic Radiation (GCR) and Solar Particle Event (SPE) radiation. GCR comes from outside the solar system and is a continuous, omni-directional flow of high energy, low intensity particles. These particles are composed of 85% protons, 13% alpha particles, and 2% heavier nuclei [5]. SPE radiation is not continual. The level of exposure from the radiation from the SPEs is dangerously high only about seven times a year when solar flares occur. During a solar flare, the Sun's chromosphere erupts and ejects high energy nucleons into space. These flares often last for time periods as long as several days. The most dangerous radiation released during SPEs is the high flux of protons [6].

The lunar radiation environment poses serious concerns for lunar base inhabitants. Proper shielding of habitation modules must be provided in order to adhere to safe radiation dose limitations. The habitation module will be an aluminum structure, and will use additional material for radiation protection. The lunar surface will provide an abundant commodity for this purpose: lunar regolith.

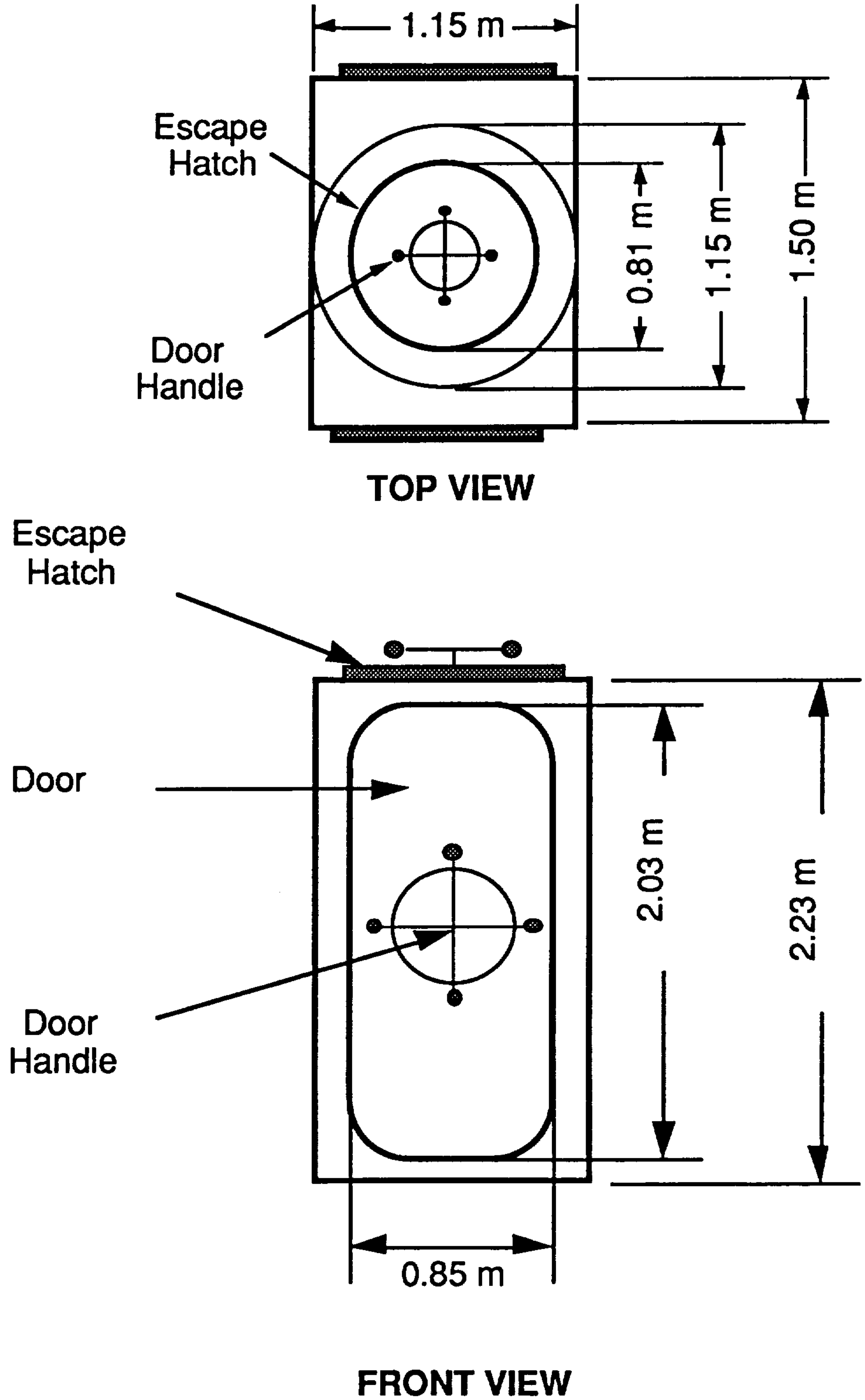


Figure 6.4. Airlock Configuration.

Lunar soil exhibits solar particle shielding characteristics similar to those of aluminum. A soil thickness between 0.3 meter and one meter is sufficient to shield against even the most energetic SPE. However, the equivalent dose from GCR increases with depth reaching a maximum in the vicinity of 0.3 meters due to the secondary effects. At depths on the order of 5 m, the equivalent dose rates approach those that are measured as background values on the Earth (0.1 REM/year). A Roentgen Equivalent Man (REM) is the amount of radiation which results due to the absorption of X-radiation. For the FLARE base habitation system, a depth of 2.4 m has been selected for burial of the lunar habitation module. This depth of shielding will yield an annual dose rate of 7.5 REM, including radiation from maximum solar flare activity [3]. Figure 6.5. illustrates a cross-sectional view of the envelope for which the decided amount of radiation protection may be provided [4].

The burial of the SSCMs to a depth of 2.4 m on the lunar surface will warrant the use of construction equipment. The procedure of burying the module will be directly related to the available excavation equipment - particularly regolith movers and cranes.

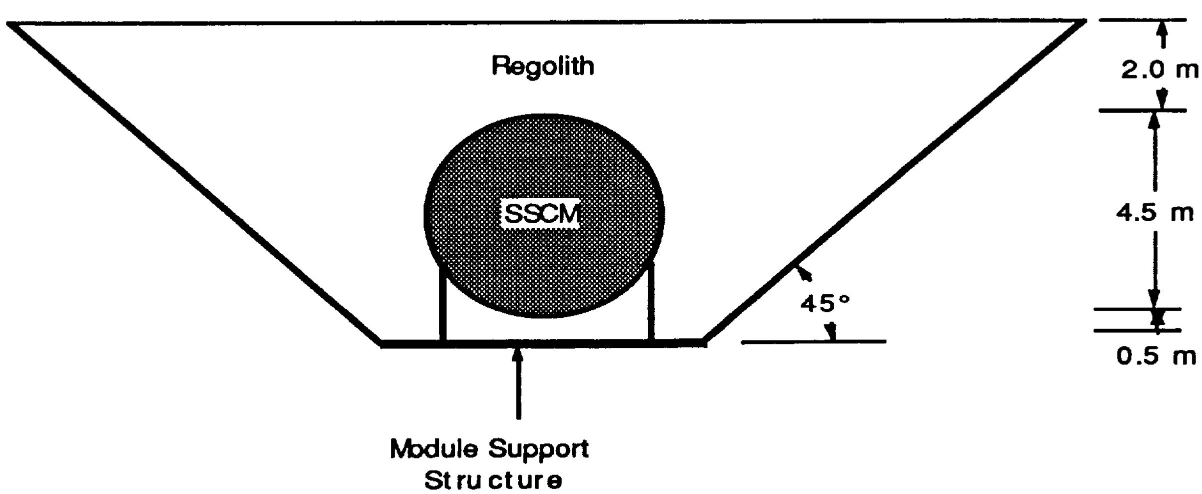


Figure 6.5. Module Burial Cross-Section.

The method recommended for burial requires excavation to a depth of seven meters with backfilling to the original lunar surface plus an additional 0.4 m of regolith placed on top. The depth of seven meters includes 0.5 m for a module structural support base of the module, 4.5 m for the outer diameter of the module, and 2 meters of lunar soil. With proper module burial techniques and reasonable soil depth, the lunar radiation environment can be managed such that human habitation is safe and feasible on the Moon.

6.6 Environmental Control and Life Support System

In order to support a man in space, a reliable life support system is required. The habitability of the lunar surface will be depend on the efficiency and redundancy of the environmental control and life support system (ECLSS). The ECLSS removes carbon dioxide, humidity, and urine from the environment, and provides water, food, and oxygen for the crew. Figure 6.6 shows a schematic of various functions of the ECLSS.

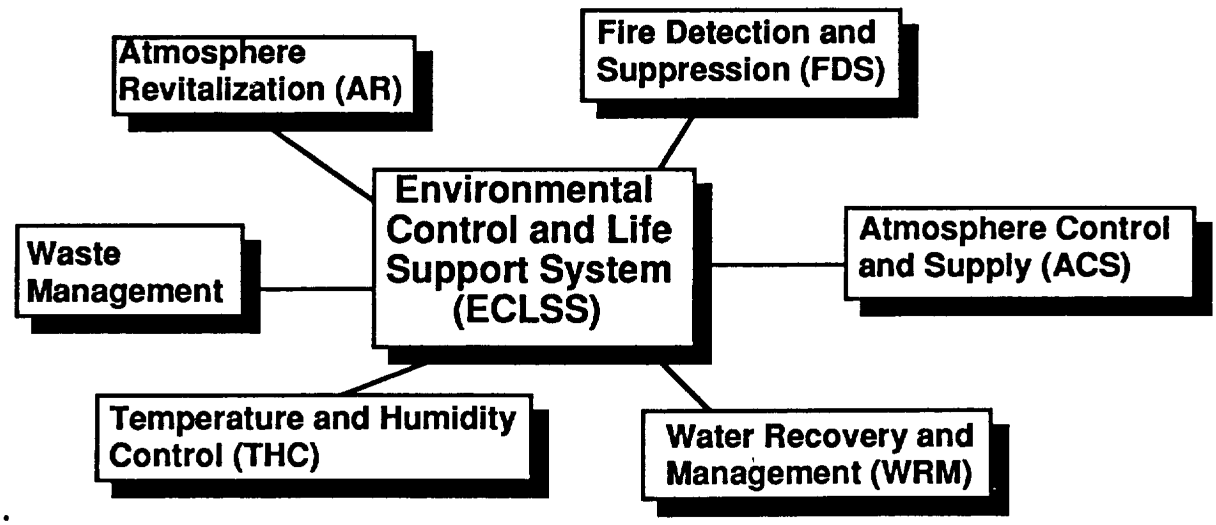


Figure 6.6. Functions of the ECLSS.

The three possible ECLSS options are the open, partially closed, and closed cycles. The open cycle ECLSS stores oxygen and water supplies. These supplies are replenished from the Earth when exhausted. The expendables needed for CO_2 removal, such as lithium hydroxide, are also stored, as is the human waste from the mission [3].

The partially closed type of ECLSS is also called the physio/chemical ECLSS. In this cycle, drinking water is recycled. Condensation from humidity and water produced by the reduction of carbon dioxide are reduced using a multifiltration procedure. The oxygen, as well as the remainder of the water requirements, is kept in storage and resupplied from Earth. The closed-cycle option incorporates regenerative subsystems that cut off the oxygen and water cycles. These subsystems will do the following:

- Remove, collect, and reduce CO_2
- Produce drinking water from humidity condensate
- Generate oxygen from water
- Reclaim cleansing water from urine and waste wash water.

Lone Star Aerospace has decided to use a partially closed cycle for the habitation module. The necessary crew provisions will be food, oxygen, and water. It is assumed that food will remain a storage-and-resupply commodity. In this partially closed system, there will be three steps in the process to convert the CO_2 to oxygen:

1. Removal and concentration of CO_2 from the cabin air
2. Reduction of CO_2 to water
3. Electrolysis of the water to produce oxygen for consumption by the crew.

The water needs of the crew will be based on water recovery processes consisting of both distillation and filtration. Contaminants of the waste water will be removed using phase change techniques., while the filtration techniques will use filters and ion-exchange resin beds for contaminant removal. Three types of water will be part of the recovery process: urine/waste water, wash water, and condensate. The partially-closed cycle will not distill the urine/waste water, but will rely on stored water. However, it will recover the condensate and wash water through a multifiltration (MF) procedure. The closed-cycle option will also incorporate multifiltration in addition to vapor compression distillation (VCD) for the urine/waste water recovery.

The crew metabolic loads will be assumed to be 2200 to 2800 calories-per-man-per-day. Estimates for water use, food intake, and associated liquid and solid wastes are shown in Table 6.3 [7]. These figures reflect loads currently used to support manned space flight, and it is projected that they will not change substantially to support a manned presence on the lunar surface.

Table 6.3. Oxygen, Water, Food Exchange.

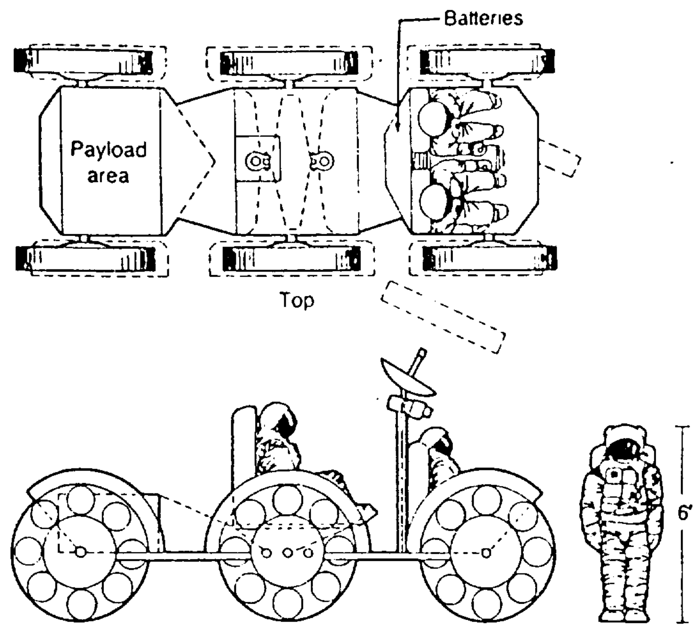
Input (lbs/day)		Output (lbs/day)	
Oxygen	22.7	Oxygen	22.7
Water	4.7	Water Vapor	2.2
Food	1.3	Urine	3.0
		Carbon Dioxide	2.2
		Solid Waste	0.6
TOTAL	**28.7**	**TOTAL**	**28.7**

6.7 Transportation

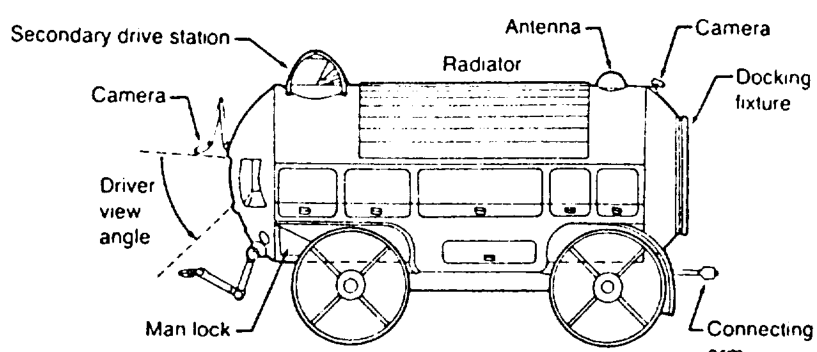
Lunar transportation for both the robotic workers and the manned observatory crew have also been investigated. Two of the lunar rovers being considered are the Local Transportation Vehicle (LOTRAN) and the Mobile Surface Application Traverse Vehicle (MOSAP).

LOTRAN is an unpressurized lunar vehicle with a range of 100 km. It has a maximum speed of 15 km per hour, and can carry two crew members [1]. LOTRAN is capable of supporting a 850 kg payload.

MOSAP, on the other hand, is a pressurized vehicle with a four piece modular design. It has a speed of 10 km per hour and has a longer range than the MOSAP (3000 km). Figures 6.7 shows both the LOTRAN and MOSAP transportation vehicles.



Local Transportation Vehicle, Unpressurized (LOTRAN)



Mobile Surface Application Traverse Vehicle, Pressurized (MOSAP)

Figure 6.7. LOTRAN and MOSAP Transportation Vehicles.

6.8 Habitation and Transportation Subsystem Overview

Lone Star Aerospace considered two modules for the habitation section: the inflatable module (IM) and the space station common module (SSCM). The structural integrity, mass, expandability, assembly, design flexibility, transportation, living space, and storage space were investigated. After analyzing the performances of the two modules in these areas , it was determined that using two SSCMs would best suit the habitation needs of the FLARE base.

These SSCMs will be cylindrical, with an inside length of 11.79 meters. This will give a total compartment volume of 58.35 m^3. Lone Star Aerospace decided to arrange these two SSCMs such that they are placed end to end and connected by a node. Two airlocks will be placed on either end of this configuration. These airlocks will provide an airtight seal, and their status will be continuously monitored by the habitat computer. To prevent any dust from entering the airlocks, a room will be placed outside each to neutralize any garment static charge buildup that might attract dust.

The habitat will be exposed to two kinds of radiation: Galactic Cosmic Radiation (GCR) and Solar Particle Events (SPE). To shield the crew against this radiation, the SSCMs will be buried and covered by lunar regolith. A depth of 2.4 meters will be necessary to provide adequate shielding.

The habitability of the lunar surface will depend on the adequacy of the environmental control and life support system (ECLSS). It will remove carbon dioxide, humidity, and urine from the environment, and provide food, water, and oxygen for the crew. Lone Star Aerospace has decided to use a partially closed ECLSS, in which the drinking water will be recycled. However, the oxygen and the rest of the water will be stored and resupplied from Earth.

For transportation, Lone Star Aerospace has decided to use two lunar rovers: one for short ranges and one for longer ranges. The Local Transportation Vehicle (LOTRAN) is unpressurized and has a range of 100 km; the Mobile Surface Application Traverse Vehicle (MOSAP) is pressurized and has a range of 3000 km.

7.0 Spacecraft Design

The following sections detail the vehicle selected to transfer the equipment necessary to set up a far side lunar observatory as well as the two potential lunar lander configurations that could provide transportation of materials from Low Lunar Orbit to the lunar surface.

7.1 Orbital Transfer Vehicle

Due to the large payloads associated with the construction of a far side lunar observatory, an Orbital Transfer Vehicle (OTV) with a very large payload capacity is required. As this requirement is beyond the range of all existing technologies for transfer vehicles, Lone Star Aerospace has selected a configuration designed by Paul Phillips of Eagle Engineering. This configuration utilizes nuclear electric propulsion (magnetoplasmadynamic ion thrusters) with a specific impulse (Isp) of 5000 sec [1]. This vehicle uses mercury as the propellant. Two hundred 30 cm mercury thrusters are used to provide a thrust of approximately 100 N. The dry weight of the vehicle is 52.7 metric tons with a total propellant weight of 29.7 metric tons. The total payload capacity of this configuration is 80.4 metric tons.

Lone Star Aerospace recognizes the dangers inherent in the utilization of mercury Therefore, the OTV will be situated such that the ion stream will not impinge on the Earth's atmosphere during the flight of the OTV to the Moon The OTV will launch from a nuclear-safe altitude of 800 km. Figure 7.1 shows a depiction of an artist's conception of the NEP freighter.

7.2 Lunar Lander Options

Two options have been studied in considering the lander that will deliver payloads from Low Lunar Orbit (LLO) to the Lunar surface.. These two options include using either a single reusable lander which will refuel from the OTV and shuttle payloads from the OTV to the lunar surface or using six decent-only lunar landers, each equipped with its own payload.

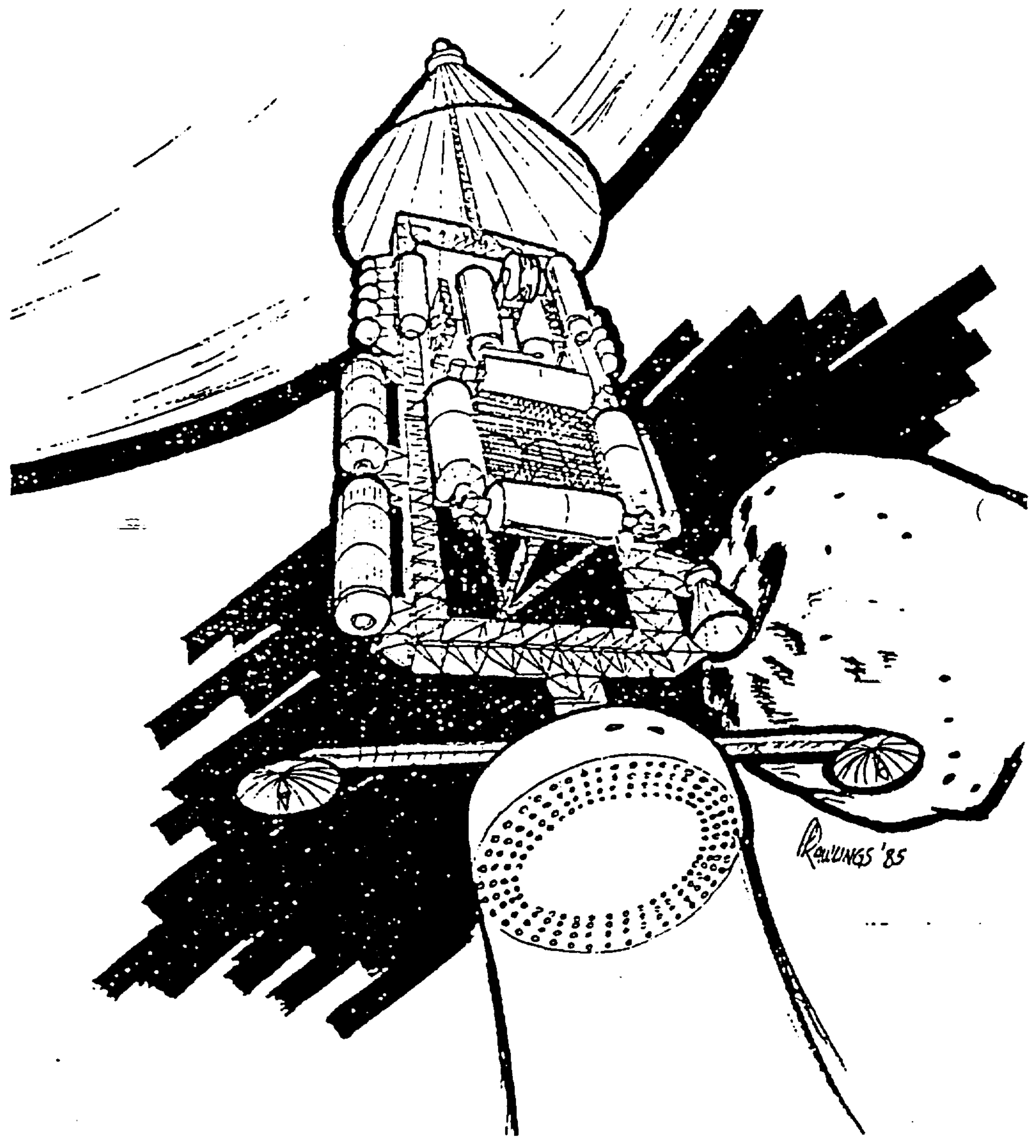


Figure 7.1. Artist's Concept of NEP [1].

7.2.1 Reusable Lander

The first lunar lander option employs the Self-Unloading Reusable Lunar Lander, a concept developed by the B&T Engineering group at the University of Texas during the Fall of 1990 [2]. This option uses a single, reusable lander that will travel from LLO to the lunar surface and back again. The lander docks with the OTV, loads enough propellant for the descent, and picks up a new payload. Using this configuration, it would require six OTV trips to accommodate all of the base instruments and supplies needed for the construction phase. Two payload/propellant combinations would be taken per OTV trip (with the exception of the first trip which would take one payload/propellant combination, the lander, and the necessary communication equipment to establish continuous contact with Earth). A sketch of the reusable lunar lander is shown is Figure 7.2.

The reusable lunar lander has a dry mass of 9.8 metric tons, a propellant capacity of 24.6 metric tons, and a payload mass of 15 metric tons. The lander uses LOX/LH2 as the descent/ascent propellant in a 6:1 oxidizer to fuel ratio with an Isp of 450 sec.

7.2.2 Lunar Operations Vehicle

The second lander configuration option involves a "one shot" lander (descent only). The Lunar Operations Vehicle (LOV) is capable of carrying 31 metric tons of payload to the lunar surface with a propellant mass of 44 metric tons and a dry weight of 5.2 metric tons [3]. The LOV would also require six OTV trips to satisfy the total lunar construction needs. One lander and one payload would be carried per OTV trip. Like the reusable lander, this vehicle uses LOX/LH2 for the propellant, but with a mixture ratio of 7:1 and an Isp of 452 sec. A sketch of the LOV can be found in Figure 7.3. A payload mass schedule was created for the LOV. This schedule is shown in Table 7.2.

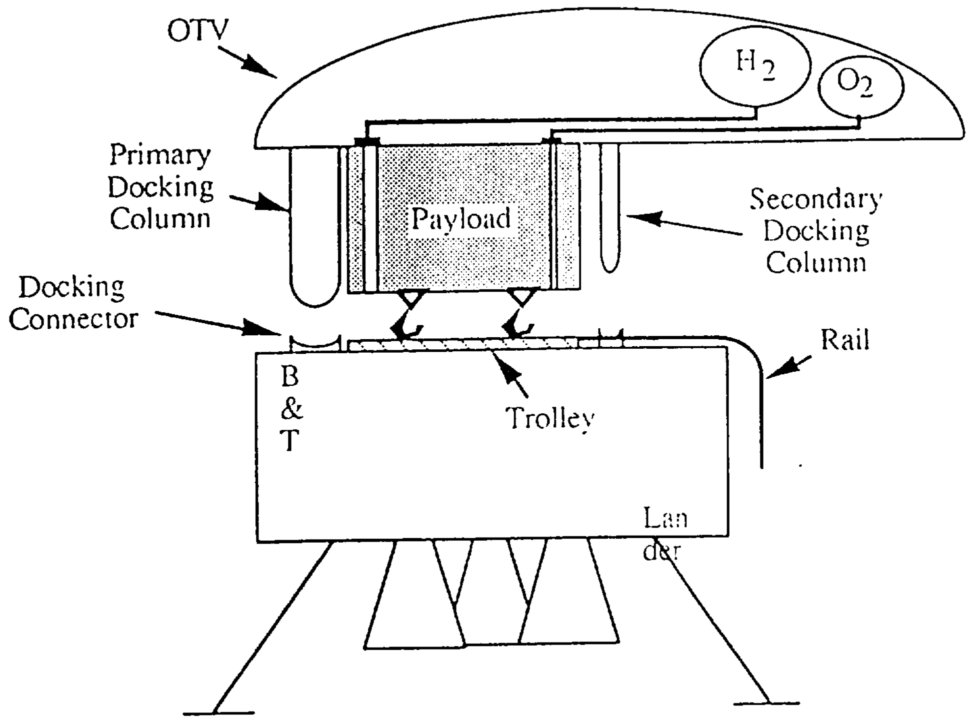


Figure 7.2. Reusable Lunar Lander [2].

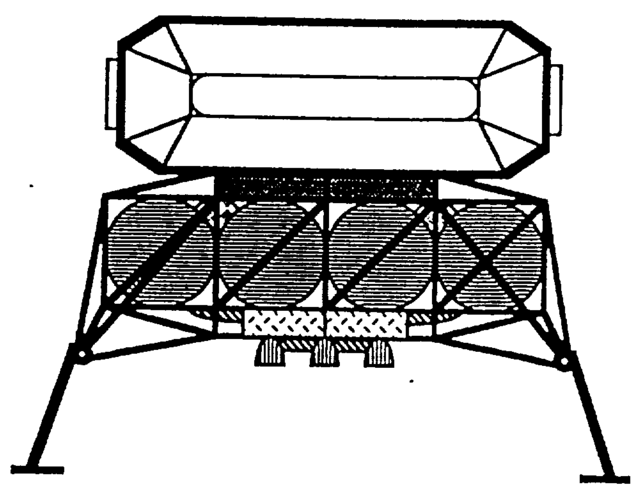


Figure 7.3. Lunar Operations Vehicle [3].

7.2.3 Lander Selection Trade Studies

To select the optimal lander for the FLARE mission, several trade studies were conducted. They are detailed below:

- Cost. The reusable lander concept would need only one lander as compared to the options of using six LOVs. However, additional technology is required for the reusable lander for docking and propellant transfer maneuvers, which would mean a higher lander cost.
- Utility. Since the design of the LOV consists of many structural members, it may be advantageous to use the LOV option. After an LOV has landed, its parts can be disassembled and incorporated into the materials to be used to construct the base.
- Feasibility. Due to the large difference in the level of technology between the two design options, it was concluded that using the LOVs would be a more feasible lunar lander option.
- Availability. Since one of the design parameters of L.S.A.'s FLARE mission is a project initialization near the turn of the century, the reusable lander option may not exist in time for this mission.

Table 7.1 shows the decision matrix which helped to determine the type of lander(s) to be used in the design of the FLARE base. From this table, it can be seen that the LOV lander option would best suit the need of the FLARE base. Table 7.2 details a possible mass schedule for the delivery of payloads using the LOV option.

Table 7.1. Lander Decision Matrix.

Alternative:	LOV	Reusable Lander
Cost	6	8
Utility	8	6
Availability	9	4
Feasibility	9	4
Total	**32**	**22**

Table 7.2 Payload Mass Schedule for the LOV.

OTV Trip Number	Equipment	Mass (metric tons)
1	Environmental Survey	.1
	Magnetoscope	.01
	1 Optical Interferometer Telescope	1.3
	Transit Telescope	1.3
	Robotics	14.0
	Full Comm. Gear	1.0
	Nuclear Plant Elements	4.0
	Solar Arrays	2.0
	Cable	6.0
	Total	**29.7**
2	Miscellaneous Supp.	2.5
	Habitat (2)	19.5
	Nuclear Plant	8.0
	Total	**30.0**
3	1 Opt. Interferometer Elem. and Corr. Pack.	1.7
	MERI	2.1
	Nuclear Plant	4.0
	Habitat (3)	19.5
	VLF	2.0
	Total	**29.3**
4	4 m Telescope	9
	Correlation Package	6
	4 Submill. Elements	8
	5 OI. Elements	6.5
	Total	**29.5**

5	Two 4m Telescopes	18.0
	4 OI Elements	5.2
	3 SI Elements	6.0
	Total	**29.2**
6	4 m Telescope	9
	1 OI Element	1.3
	Gamma Ray	3.0
	X-Ray	2.0
	High Cosmic Ray Det.	3.0
	Low Cosmic Ray Det.	3.0
	Geological Package	.75
	Mat. Plants I,II, & III	1.76
	Total	**23.8**

8.0 Trajectory Analysis

The following sections describe the analysis of the Earth to Moon trajectories involved in transporting all the necessary materials from LEO to the lunar base. The purpose of this analysis is to provide the spacecraft subsystem team with delta-v information to be used in transfer vehicle sizing. Lone Star Aerospace does not attempt to precisely characterize the Earth to Moon transfer. Such an analysis is beyond the scope of this project. The following assumptions were made to accomplish the simplified analysis:

- All transfer missions start from LEO
- A transportation node such as a space station is assumed to exist in LEO to support the lunar transfer missions
- The LEO is assumed to be coplanar with the Moon's orbit about the Earth
- The only perturbing forces on the lunar transfer vehicle are the gravity forces of the Earth and the Moon
- The spacecraft has no perturbing affect on the Earth and the Moon (i.e. the spacecraft's mass is negligible).

8.1 Patched Conic Approximation

The patched conic approximation with a Hohmann transfer ellipse was used to calculate the approximate delta-v requirement for a vehicle travelling from LEO (Low Earth Orbit) to LLO (Low Lunar Orbit). The geometry of this type of transfer is shown in Figure 8.1. This approximation uses defined spheres of influence for the Earth and Moon. While in the sphere of influence of a body, the spacecraft is assumed to be affected only by the gravity of that body. Hence, two-body motion can be used to calculate the delta-v requirements.

LEO was defined as a 400 km altitude orbit with an inclination of 28.5 degrees, a typical space station orbit [1]. LLO was defined as a 200 km altitude orbit. Orbits below this altitude tend to degenerate rapidly and orbits above this altitude experience fluctuations in eccentricity due to the perturbations of the Sun and the oblateness of the Earth [2].

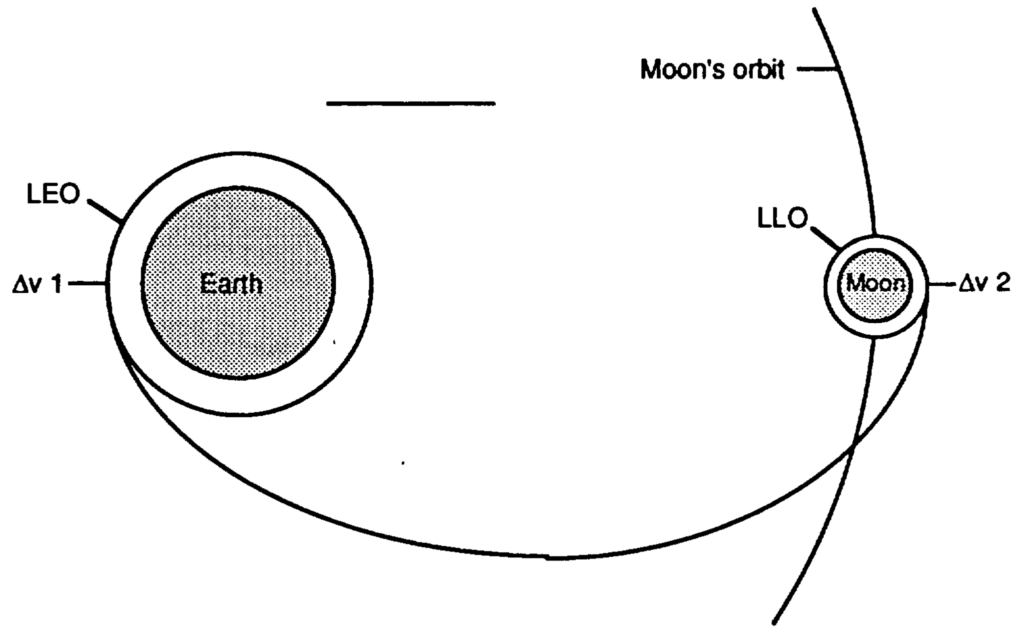


Figure 8.1. Earth to Moon Hohmann Transfer Ellipse.

The delta-v needed to leave LEO and inject into the lunar transfer orbit was calculated to be approximately 3.1 km/s. A delta-v of .829 km/s was calculated to enter LLO from the lunar transfer orbit. This results in a total delta-v from LEO to LLO of approximately 3.9 km/s. Calculations of the delta-v requirements can be found in Appendix F. The typical delta-v for Earth to Moon transfers is approximately 4 km/s [3]. The time of flight for this lunar transfer is approximately five days.

Since the Moon's orbit varies between 18°19' and 28°35' in inclination with respect to the Earth's equator (with a period of about 18.6 years), it is likely that a lunar transfer will require a plane change delta-v in addition to the delta-v of approximately 4 km/s [4]. Therefore, this calculated delta-v was taken as a minimum for lunar transfer and given to the spacecraft subsystem team for initial vehicle sizing. Using this delta-v, the spacecraft subsystem team determined that an ion propulsion transfer vehicle would be more efficient than a chemical propulsion system, since it would use less than half the amount of fuel that a chemical propulsion vehicle would use. Therefore, an ion propulsion vehicle was chosen as the lunar transfer cargo vehicle for this mission.

Trajectory analysis methods such as the patched conics approximation cannot be used to characterize the trajectory of an ion propulsion vehicle because these methods assume instantaneous delta-v impulses. Ion propulsion systems cannot provide short impulses of high thrust as can chemical propulsion systems. Instead, they achieve the required delta-v by providing low thrust over a long period of time.

Characterization of a low thrust orbit requires the integration of the equations of motion using the restricted three body approximation. Due to the time constraint under which this project was undertaken, an analysis of this magnitude was not possible. Instead, a previous analysis of the low thrust trajectory of a lunar transfer vehicle was used. The following is a brief description of that analysis.

8.2 Low Thrust Trajectory

The use of ion propulsion necessitates additional assumptions for the mission scenario. Since an ion propulsion system will most likely be nuclear powered, the transfer vehicle's trajectory will be restricted by a proposed Nuclear Safe Orbit (NSO). Such an orbit is assumed to be defined as an 800 km altitude orbit. Furthermore. nuclear power in space is assumed to be acceptable at the time of this mission [3]. The vehicle may not enter an altitude below this 800 km altitude orbit at any point in the trajectory. Also, since the propellant for the OTV (Orbital Transfer Vehicle) is mercury, the vehicle cannot thrust in the direction of Earth. Avoiding this geometry is feasible since the low thrust trajectories are optimal when the thrusting vector is tangent to the vehicle trajectory [5].

Using a low thrust to achieve the required delta-v, the vehicle will spiral outward from Earth orbit and spiral into low lunar orbit. Figures 8.2 and 8.3 show the Earth to Moon trajectory and the Moon to Earth trajectory, respectively. These trajectories are shown with respect to the geocentric, rotating coordinate system in which the primary direction is always from the Earth to the Moon (see Figure 8.4). Note that Figure 8.2 shows a retrograde lunar capture. This is not necessarily a mission characteristic. This figure is used merely to illustrate the geometry of the spiral trajectory. The perturbing effects of the Earth's and the Moon's gravity become more significant for these slow spirals, hence conic sections cannot be used to properly characterize the trajectory. Instead, a restricted three body formulation must be used.

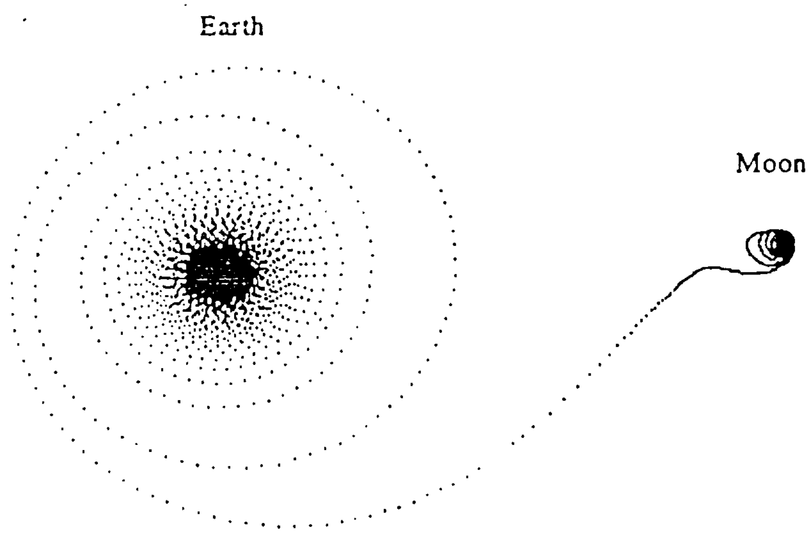


Figure 8.2. Typical Earth to Moon Trajectory for Low Thrust Vehicle [3].

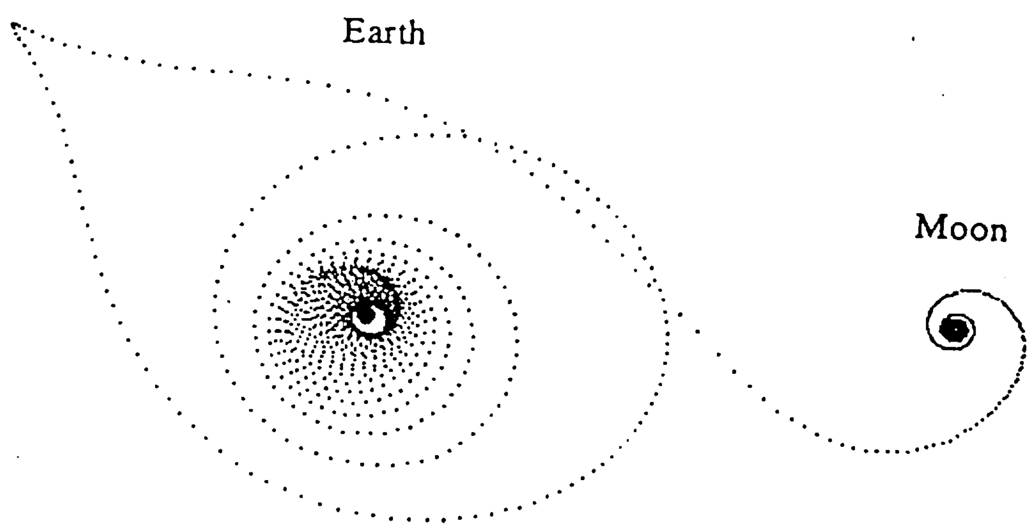


Figure 8.3. Typical Moon to Earth Trajectory for a Low Thrust Vehicle [3].

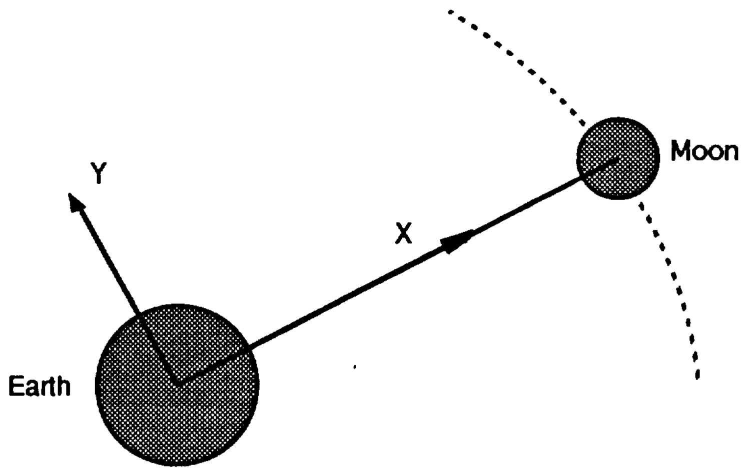


Figure 8.4. Earth-Moon Rotating Coordinate System

The trajectory is divided into three phases: departure, trans-lunar targeting, and capture. In the first phase, the vehicle applies a thrust tangential to its orbit to spiral away from the Earth. During the trans-lunar phase, the value of the Jacobian constant is calculated and compared to the required value for a successful trans-lunar trajectory. This comparison determines the thrust required. For the final capture phase, the velocity of the vehicle relative to the Moon is compared to a pre-calculated velocity profile for a spiral capture trajectory, and the appropriate corrective thrust is applied [3]. A case study using a Nuclear Electric Propulsion (NEP) vehicle was performed by David Korsmeyer of the University of Texas. Table 8.1 shows the resulting data. This table shows the required initial mass (Mo), the payload capability, propellant mass (Mp), the time of flight, and the radius of the final orbit.

The ion propulsion vehicle is very efficient for large cargo transfer, but the long times of flight make the use of this vehicle impossible for human crew transportation. Instead, chemical propulsion spacecrafts similar to the Apollo series of spacecrafts must be used. The trajectory used for these manned missions is briefly described in the next section.

Table 8.1. Trajectory Results for the NEP Vehicle [3].

	Mo	Payload	Mp	Time Of Flight	Final Radius
Earth to Moon	154 MT	80.9 MT	21.4 MT	130 days	1865 km
Moon to Earth	60 MT	8.3 MT	13.7 MT	82.82 days	7177 km

8.3 Manned Mission Trajectory

For manned missions to the Moon, safety is the most important consideration. If the spacecraft experiences a malfunction during the translunar phase of the Earth-Moon trajectory or is unable to insert into lunar orbit, the vehicle must have the ability to return to Earth without a large delta-v maneuver. Manned lunar mission trajectories are designed to include this contingency. The trajectories used are referred to as hybrid free-return trajectories because some mid-course corrections are required to achieve the return to Earth. The geometry of the free-return trajectory is shown in Figure 8.5. This choice of trajectory for manned missions exhibited its usefulness during the Apollo 13 mission. A system malfunction after translunar injection forced the free-return contingency to be used, which prevented the loss of life.

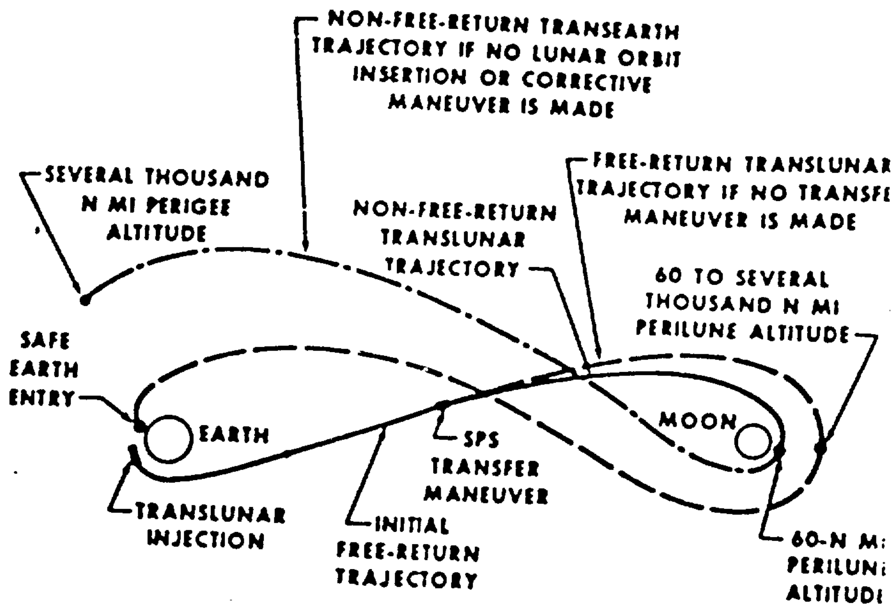


Figure 8.5. Hybrid Free-Return Trajectory [6].

8.5 Trajectory Analysis Overview

A total delta-v of approximately 3.9 km/s was calculated for a Hohmann transfer from a 400 km LEO to a 200 km LLO. With this delta-v, the spacecraft subsystem team found that an ion propulsion vehicle would be the best choice of OTV. This OTV will use a low thrust over long periods of time to achieve the required delta-v. The resulting trajectory spirals away from the Earth and spirals into lunar orbit.

This vehicle will not be able to transfer human crew to the Moon, however, because of the long times of flight for transfer (approximately 130 days from the Earth to the Moon). Instead, for manned missions, an Apollo type spacecraft will be used. This vehicle will use a hybrid free-return trajectory which allows the safe return of the vehicle to Earth orbit in the event of a malfunction.

9.0 Thermal Control Systems

The purpose of a Thermal Control System (TCS) is to keep the temperature of the base and various subsystems within established limits. The TCS cools or heats the desired location through interface heat exchangers, transports heat from sources to sinks, and rejects heat by various methods.

Thermal control systems can be divided into two types: active and passive. An Active Thermal Control System (ATCS) involves the operation of mechanical or electrical devices (fluid pumps, resistance heaters, etc.) to maintain desired temperature. A Passive Thermal Control System (PTCS) relies on conduction and radiation alone to transport heat. A PTCS will be provided by the abundant lunar regolith and manufactured shading for some instrumentation elements. Lunar regolith is an excellent insulator with low thermal conductivity. Since the lunar base modules will be covered by about two meters of regolith, this will provide a stable thermal environment and will enhance the TCS design.

9.1 Active Thermal Control System

The lunar habitation facility will generate an excess heat load that must be rejected. In order to maintain thermal stability inside the module an Active Thermal Control System (ATCS) must be incorporated. The ATCS is comprised of three stages in the elimination of waste heat: acquisition, transportion, and rejection. The heat acquisition and transportion segment of the ATCS for the lunar common module are designed after the space station system. Heat rejection analyses involve three variations of heat-pipe radiators on the lunar surface.

9.1.1 Thermal Bus

Thermal control requirements for future space applications are becoming increasingly more stringent with respect to temperature control, quantity of waste heat to be rejected, and transport distance. Conventional single-phase, pumped liquid technologies are inadequate for several reasons. Small-scale applications typically cannot provide the narrow temperature control ranges required, and larger scale applications tend to be expensive with respect to weight and power requirements. Two-phase loop concepts can potentially satisfy all currently identified operating requirements with good adaptability, versatility, and low weight and power penalties. Advantages of the two-phase fluids include an enhanced thermal capacity, improved heat transfer coefficients, decreased pumping power requirements, and reduced fluid inventory.

The central element of the ATCS is a thermal bus, which provides the function of waste heat acquisition and transport for the system. The thermal bus, as adopted from space station designs, consists of a two-phase loop system that supports a uniform thermal control source (cooling or heating), interfaces for all heat loads and heat rejection system. The thermal bus is separated into two loop systems: 1) a two-phase ammonia transport loop, external to the habitation modules, which transports the heat load from the modules to the radiator for heat rejection, and 2) a two-phase water loop, internal to the habitation modules, which collects the waste heat from the equipment coldplates and transports it to the module/external thermal bus interface heat exchangers.

The external and internal thermal transport loops for the habitation module are segmented into two separate temperature levels, 2° C and 21° C, to accommodate certain equipment standards and to enhance the isothermal characteristics of the thermal management system [1]. The 2° C loop services the ECLSS and personal hygiene loads, while the 21° C services the equipment for the galley, workstation, and health maintenance facility.

The internal system consists of a closed fluid loop maintained in circulation by a pump. The working fluid, water, is preconditioned in a supply reservoir to be near saturation at the designated temperature level to provide the heat source or sink necessary for the temperature control of the equipment. A two-phase saturated mixture of the working fluid flows through heat stations arranged in parallel along the flow loop. Heat exchange with the equipment involves change of phase, either condensation or evaporation, depending on whether the various items require cooling or heating [2].

The external thermal bus, a two-phase ammonia loop, is required if radiators are used for the heat rejection system. The ammonia loop interfaces with the internal thermal bus through heat exchangers. Liquid ammonia is pumped into a supply header which feeds individual heat exchangers. A control valve at the entrance to the exchanger admits the pressurized liquid in response to a sensor that determines the amount of liquid remaining in the exchanger. The waste heat collected by the internal thermal bus vaporizes the ammonia. Vapor exits the exchanger to the vapor collection header and is transported to the condenser. The radiator, in conjunction with the condenser, removes the heat leaving the ammonia liquified and subcooled before being pumped to the heat sources to continue the cycle. Thermal storage devices are included in the external bus system for heat load leveling, which allows accommodation of peak heat loads and variation of radiator heat rejection capacity [1].

Figure 9.1 shows a schematic of the external thermal bus management system. The key stages along the bus line include the ammonia/water loop heat exchanger, bus condenser, space radiator, accumulator, and pump. The driving design parameter of the component characteristic is the type of working fluid. Ammonia was selected as the working fluid due to its attractive thermal factors, including a large heat of vaporization, low freezing temperature, and high specific heat. In a comparison with Freon, the two-phase ammonia requires significantly less pumping power, lower total weight, and smaller line sizes [3].

9.1.2 Heat Rejection

The final process of the ATCS is the heat rejection. A variety of heat rejection processes were studied for this project: liquid droplet radiators, conduction to soil, convection to water tank, and surface radiators. After a preliminary assessment of the options, the surface radiator emerged as the most viable for further analysis.

The radiator configuration selected for the habitation facilities consists of high capacity, monogroove, heat-pipe radiator panels. Figure 9.2 shows the components. The high capacity is due to the separate, low pressure drop liquid flow channel. Other advantages include its light weight and independence from variable conductance controls and ion pumps. The radiator is comprised of multiple interlocking panels that plug into a contact heat exchanger. The ammonia thermal bus transports the habitation module waste heat to the contact heat exchanger which interfaces with the radiator.

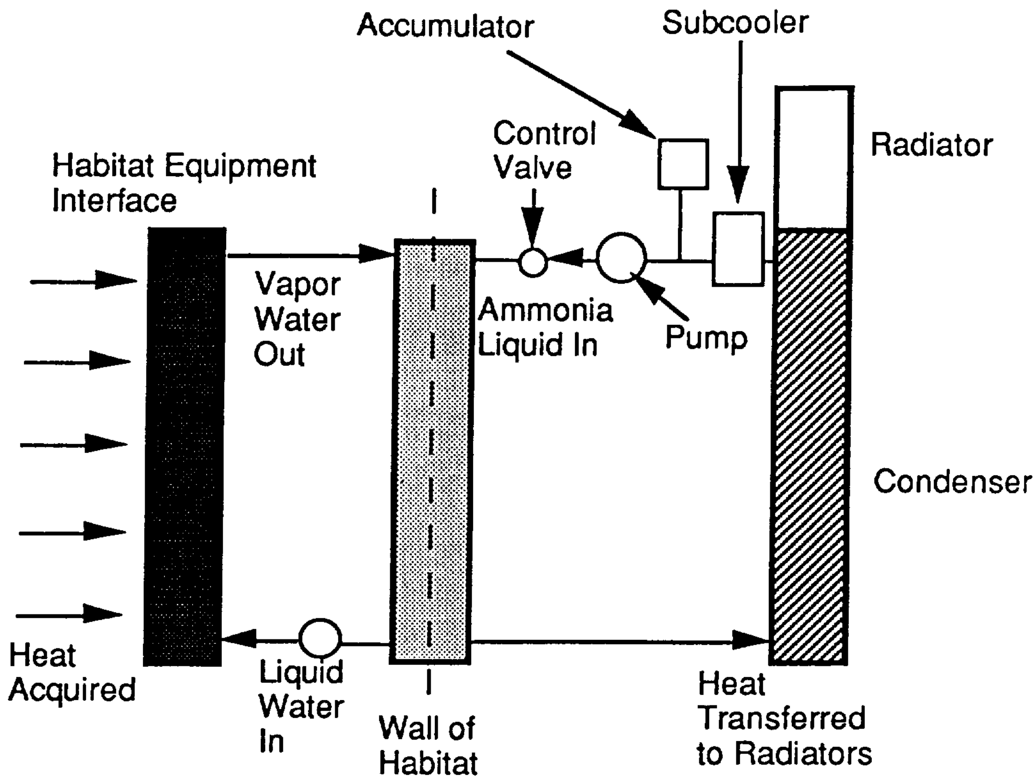


Figure 9.1. Schematic of External Bus Management System.

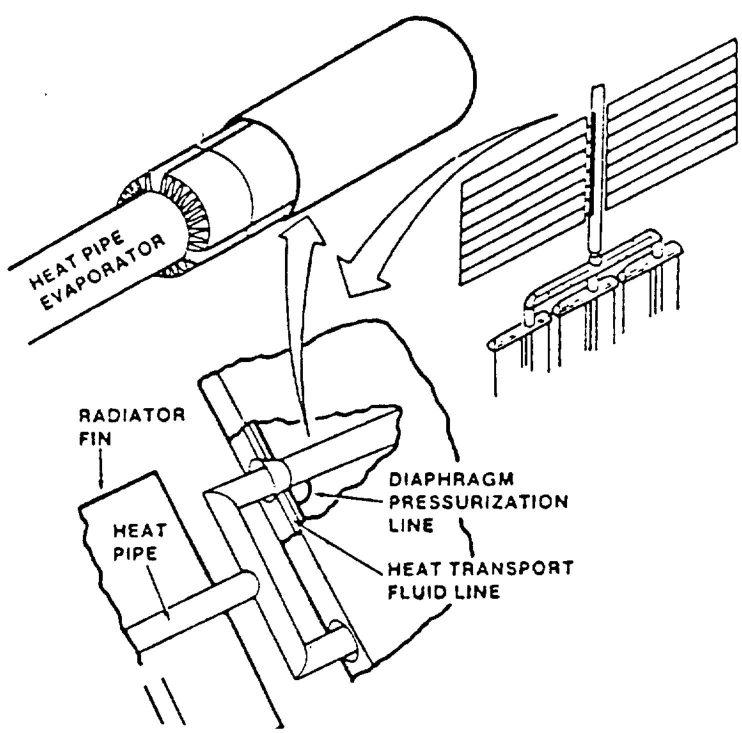


Figure 9.2. Heat-Pipe Radiator Configuration [2].

The lunar environmental properties, particularly the radiation emitted by the lunar surface, contribute to the total amount of heat the radiator must reject. In addition to the waste heat of the habitation modules, other excess heat sources include solar radiation and view factor effects. In order to properly size the surface radiator, solar flux is determined using the full solar constant of 1390 w/m^2. View factor effects concern the fraction of the radiation emitted by the lunar surface which is intercepted by the radiator. For this analysis, the radiator and lunar surface are modeled as perpendicular rectangles with a common edge, resulting in a view factor of 0.9.

The irradiation calculations as well as the emissive power of the radiator depend on specific radiator characteristics. Adopted from heat-pipe radiator data for the space station, the absorptivity is selected at 0.2 and the emissivity at 0.9. The emissive power of the radiator, or the rate of radiant energy emitted by the radiator surface, depends upon the working fluid temperature of the thermal bus. Therefore, two separate radiator sizings are needed, one for the 2°C loop and one for the 21°C loop.

Three different radiator configurations are analyzed within the sizing algorithm. The variations pertain to the extent of the radiator's exposure to the environmental irradiation factors, during the Moon's day cycle. The three versions include the radiator at full view, rotating, and with a canopy. The full view configuration implies the worst case in which the radiator surface directly receives total solar flux. The rotating option places the radiator on a gimbal system so that the radiator can be preferentially oriented to obtain low effective irradiation. The gimbal system is composed of two main parts: A rotary coupling to accomplish the rotation of the radiator, and a sensor/control system to detect the minimum thermal environment. The rotating version reduces the solar constant by 50%. The canopy option provides the radiator with a sun shield. The canopy is tilted to continuously cast a shadow on the heat-pipe radiator. The top of the sunshield would be coated with the most efficient white oxide (lowest absorptivity/emissivity ratio). The lower side would be anodized aluminum that would reduce the reflection of the lunar surface heat. Such a canopy configuration would decrease the solar constant by 65%, the lunar radiation by 50%, and the view factor effects by 40%. Table 9.1 summarizes the three radiator options and their corresponding irradiation flux values [1].

Table 9.1. Environmental Effects on Three Versions of the Heat-Pipe Radiator.

Environmental Inputs	Full View	Rotating	Canopy
Solar Constant, w/m^2	1390	695	486.5
Lunar Radiation, w/m^2	1100	1100	550
View Factor	0.9	.9	.55
Additional Waste Heat			
Solar Irradiation, w/m^2	278	139	97.3
Lunar Irradiation, w/m^2	220	220	110
View Factor Effects	250.82	250.82	153.28

Although the irradiation flux changes for the three options, the emissive power of each of the radiators is the same regardless of the radiator configuration. The emissive powers for the 21°C loop and the 2°C loop are 381.25 w/m^2 and 291.85 w/m^2, respectively. Calculation of the radiator area involves the total irradiation factors, the emissive power, and the habitation module waste heat. The 21°C radiator facilitates the removal of excess energy from the crew quarters, the galley/wardroom, the workstation, and the health maintenance area. For a crew of six persons, the amount of heat to be rejected is approximately 16.82 kW. Associated radiator sizes appear in Table 9.2.

Table 9.2. 21°C Radiator Sizing for Three Versions of the Heat-Pipe Radiator.

21°C Radiator Sizing	Full View	Rotating	Canopy
Total Radiation Area, m^2	1230.75	110.18	41.86
Effective Area, m^2	615.37	55.09	20.93
Radiator Mass, kg	8412.66	753.15	286.10
Radiator Volume, m^3	375.13	33.58	12.76

The total radiating area refers to the total surface area needed for proper heat rejection. The radiator is actually double-sided thus reducing the radiator area in half. The full view radiator configuration appears massive at 615.37 m^2. The rotating version is more reasonably sized at 55.09 m^2, while the radiator with canopy reduces the necessary are to 20.93 m^2. The radiator mass values are based on space station estimates for heat-pipe radiators which project a weight/area ratio of 6.83 kg/m^2 [3]. Resulting masses follow the same trend as the areas, with the sun-shielded radiator weighing the least. The volume sizing is based on the average thickness of heat-pipe radiators, approximately 0.305 m.

The 2°C radiator facilitates the removal of excess energy from the ECLSS and the personal hygiene area. For a crew of six persons the amount of heat to be rejected is 4.21 kw. Table 9.3 details the resulting areas, masses, and volumes for the three radiator configurations at this lower rejection temperature.

Table 9.3. 2°C Radiator Sizing for Three Versions of the Heat-Pipe Radiator.

2°C Radiator Sizing	Full View	Rotating	Canopy
Total Radiation Area, m^2	-25.48	-161.01	18.86
Effective Area, m^2	-12.74	-80.50	9.43
Radiator Mass, kg	impossible	impossible	128.92
Radiator Volume, m^3	impossible	impossible	5.75

Calculations show that the full view and rotating versions are not capable of expelling the heat absorbed by the radiator. The effectiveness of a heat radiator is determined by the net heat dissipated, *i.e.*, the difference between the heat radiated and the heat absorbed. The low emissive power associated with the 2°C temperature results in a negative radiator effectiveness in which the heat absorbed is greater than the heat radiated. Therefore, under these conditions the full view and the rotating radiators are not feasible for 2°C thermal loop heat rejection. The canopy configuration provides adequate shielding from the incoming irradiation fluxes. For the radiator with canopy, the total radiating area equals 18.86 m^2 with a resulting radiator size of 9.43 m^2. The radiator mass, 128.92 kg, and volume, 5.75 m^3, are calculated as prescribed for the 21°C radiator.

The selection of radiator configuration for the heat rejection system depends on the loop temperature. For the 21°C loop all three options are possible; however, the canopy version is the only option for the 2°C loop. The canopy design incurs additional considerations, such as the mass of the canopy and the construction and assemble of the configuration. The canopy also affords an additional advantage by protecting the heat-pipe radiator from the dust on the lunar surface. The full view and rotating radiators are completely subjected to the lunar dust environment which could hinder their heat rejection performance, as well as incur significant maintenance requirements. Finally, the shading canopy could also double as a shield against micro-meteoroid impacts.

10.0 Robotics

Almost every part of the lunar base will involve construction by robots. Four telerobotic rover units have been proposed to perform the tasks needed for base construction. These units include a heavy lift crane (HLC), an excavator/digger, and two smaller assembly rovers. Each rover will be equipped with some level of Artificial Intelligence (AI) which will enable it to perform specific tasks and move about the lunar surface without requiring total teleoperator control.

10.1 Heavy Lift Crane

The HLC will be used to remove payloads from landers and place them in their proper positions. Potentially, some landers could set down very close to the site where a specific payload is to be unloaded and assembled instead of using the regular landing pad. For example, the nuclear reactors will be set up three kilometers from the habitat for safety reasons. An LOV carrying the reactors could land near this site and the robotics team would then drive out to the reactor site for unloading and assembly. This system might prevent the HLC from having to traverse long distances while carrying large packages.

Two different conceptual designs for the HLC have been proposed. The first is shown in Figure 10.1. This design incorporates a long extendable boom to lift payloads off landers and large retractable outriggers for stability. This type of crane would not be very stable when moving about on the lunar surface and loaded with a heavy payload. A second design is shown in Figure 10.2. Its horseshoe shape allows the crane to drive up to the payload and lift it off the lander. Since the payload would be held over the center of the crane, this design would allow for more stable mobility while carrying a payload.

As previously stated, the packages can possibly be delivered to their approximate assembly site by the LOV, where they will be unloaded. For this reason, mobility while burdened with a payload may not necessarily be a problem. Therefore, the first of the two crane designs presented was selected.

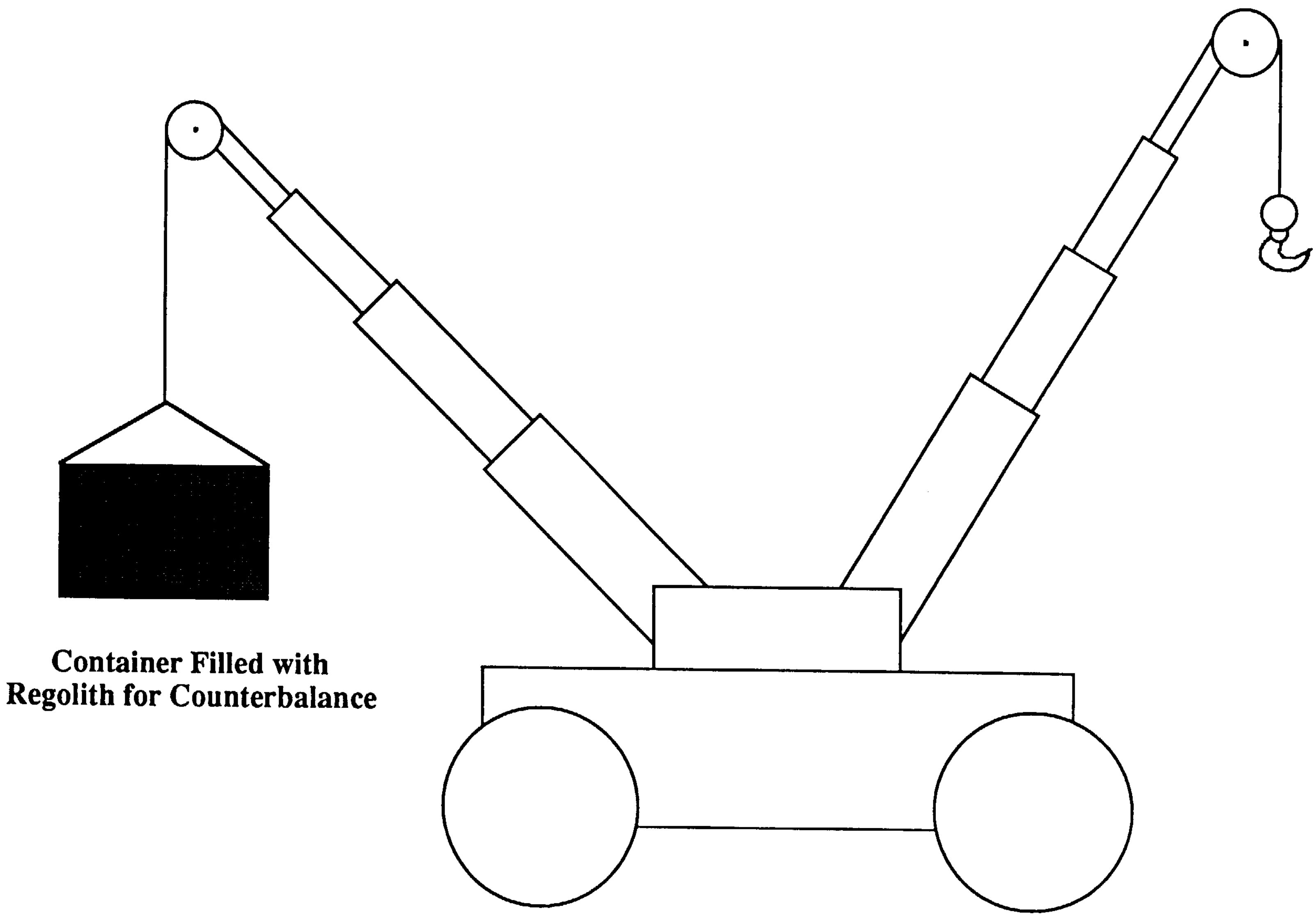


Figure 10.1. Primary Crane Design.

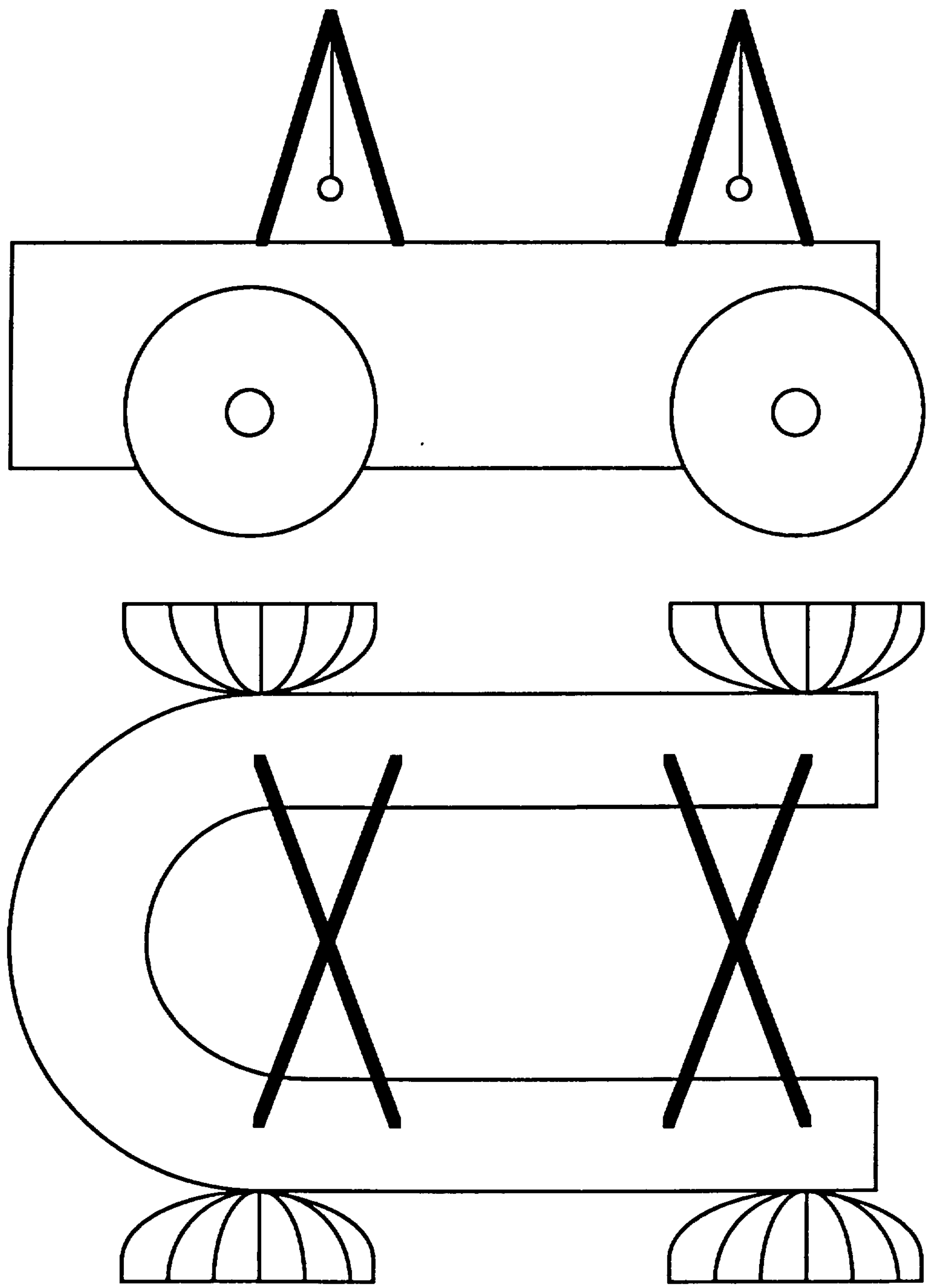


Figure 10.2. Alternate Crane Design.

10.2 Excavator/Digger

The excavator/digger will dig the holes in which the habitats, nuclear reactors and other payloads will be placed. The excavator/digger will also level areas on the lunar surface for landing pads and instruments. The excavator/digger, as shown in Figure 10.3, will resemble a bulldozer. As an experiment, the excavator/digger will also be equipped with a coring device and explosive charges. This will test the viability of blasting holes in the lunar regolith for payload placement.

The excavator/digger will also perform the task of burying power/communication cable. A trenching device mounted on the rear of the excavator/digger will dig a shallow 0.2 meter trough in which cable being fed from the excavator/digger will be laid and subsequently covered by the plow attachment. This will prevent damage to the cable from either micrometeorites or a rover's wheels.

10.3 Smaller Rovers

The smaller rovers will help set up the instruments and repair the larger rovers and each other in case of robotic system failures. All of the rovers will be constructed in such a way that majority of the on-board systems are modular and can be interchanged with new parts if one system fails. Some spare parts such as batteries and attachments will be transported on the first OTV. If other parts are needed to repair a major systems failure, they can be transported on the next OTV if space permits.

10.4 Autonomous Control

Due to the 3 second round-trip time delay in communications with the telerobotics elements, visual navigation and hazard avoidance must be incorporated into each rover to insure safe mobility from one side of the base to the next. The rovers will also need a way to determine absolute position on the lunar surface. This will be accomplished by either a system of beacon towers on the outer perimeter of the base or through utilization of the communications satellite in the L2 halo orbit.

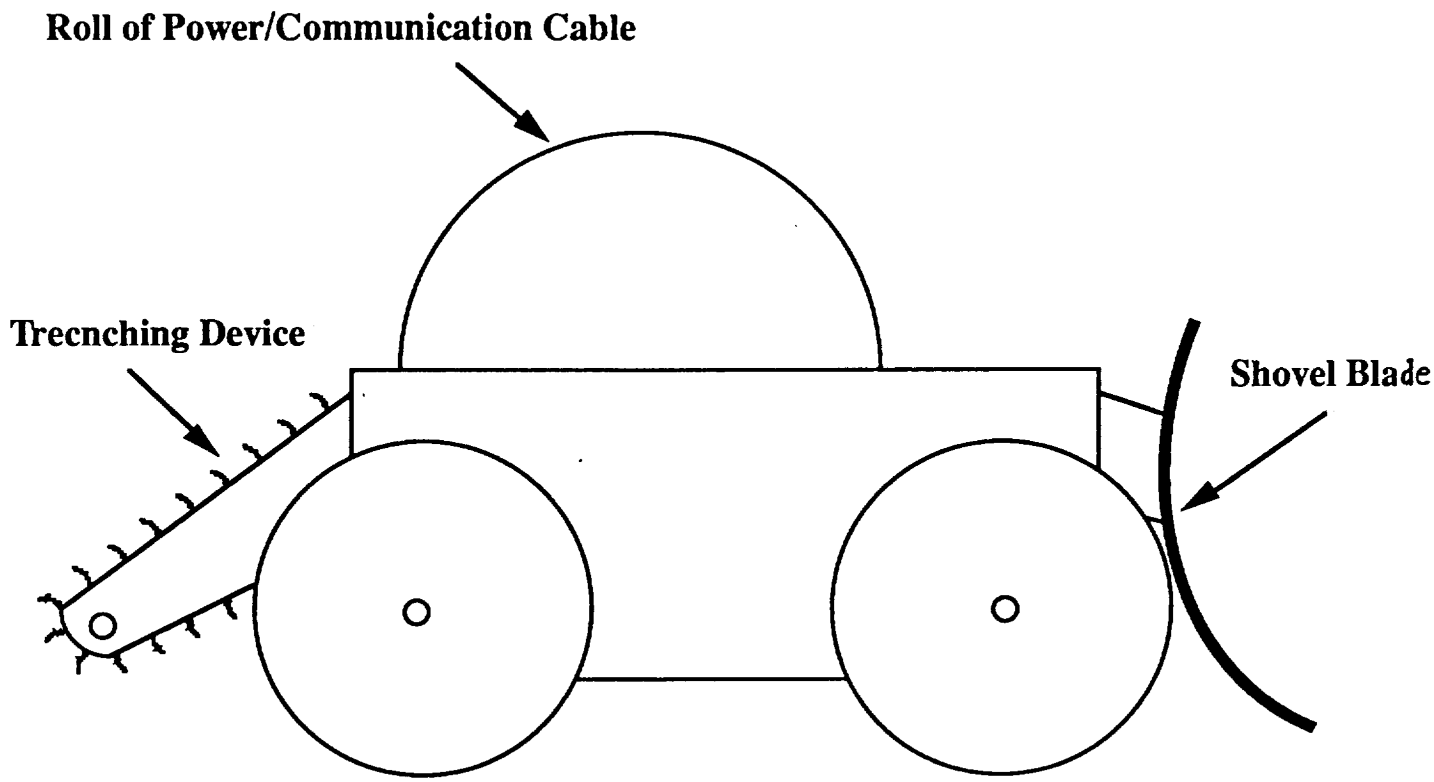


Figure 10.3. Excavator/Digger.

10.4.1 Visual Navigation and Hazard Avoidance

As a result of the time delay, complete teleoperation of the robotic rovers would be impractical and difficult. However, the technology required for a highly autonomous robot (which could roam about the lunar surface and perform tasks without any human intervention) is yet to be developed. Two different designs which use various degrees of autonomy have been proposed by the Jet Propulsion Laboratory (JPL) in Pasadena, California for a Mars type rover. The two different systems are called Computer Aided Remote Driving (CARD) and Semi-autonomous Mobility (SAM) [1].

10.4.2 Computer Aided Remote Driving

The CARD concept was originally developed by Brian Wilcox at JPL. With CARD, stereo pictures from the rover are sent to Earth where they are viewed by a human operator using a stereo display. The operator designates a path for the vehicle to follow as far ahead as he can see accurately in three dimensions. A ground based computer calculates the turn angles and path segment distances that define the route. The information is sent to the rover, which executes the path by dead reckoning (perhaps aided by computer vision). A new stereo pair of pictures is taken from the new position and the whole process repeats. Depending on the terrain, the rover might travel from 20 to 30 meters on each iteration which will take approximately 20 to 30 minutes to perform. This averages out to about one to three centimeters per second. This type of navigation would be useful in particularly difficult terrain or over very short distances. However, where long-distance mobility is needed (such as required on the lunar observatory base), the CARD method would prove to be much too slow.

10.4.3 Semi-Autonomous Mobility

The SAM method was developed by Donald Gennery, based on a suggestion from others at JPL. In the SAM method, local routes are planned autonomously using images obtained on the vehicle, but they are guided by global routes planned less frequently by humans using a topographic map obtained by a lunar orbiter. Topographic maps of the lunar surface have already been gathered; but newer, more accurate images might be needed.

The sequence of operations in the portion of SAM involving Earth is completed as described in the following section (see Figure 10.4 for visual aid). As commanded from Earth, the orbiter/communications satellite takes a stereo pair of pictures (by taking the two pictures at different points in the orbit) of an area to be traversed (if this area is not already mapped). These pictures should have a resolution of approximately one meter. The pictures are then sent to Earth, where a human operator uses them to designate an approximate path for the vehicle to follow. This path is designed to avoid large obstacles, dangerous areas, and dead-end paths. This path and topographic map for the surrounding area are then sent from Earth to the rover. This process repeats as often as needed, perhaps as often as the time necessary to travel between the different instrument packages which need tending.

The sequence of operations in the portion of SAM taking place on the lunar surface begins when the rover views the local scene. By means of a scanning laser and/or stereo cameras, the rover computes a local topographic map. This map is matched to the local portion of the global map sent from Earth, as constrained by knowledge of the rover's current position from other navigation devices or previous positions, to determine the accurate rover position and to register the local map to the global map. The local map (from the rover's sensors) and the global map (from the Earth) are then combined to form a revised map that has high resolution in the vicinity of the rover. This map is analyzed by computation on the rover to determine the safe areas over which to drive. A new path is then computed, revising the approximate path sent from Earth. With the local high resolution map, small obstacles can be seen which might have been missed in the low-resolution pictures used on Earth. Using the revised path, the rover then drives ahead a short distance (perhaps ten meters), and the process repeats. This method is a considerable improvement over the CARD method and allows for an average rate of travel of about 14 cm/sec. Also, the SAM system does not require constant communication with the rovers from Earth, as does the CARD system.

This type of mobility was designed for a Mars rover which would traverse new ground almost constantly. Since the lunar rovers used to construct and later maintain the observatory will be traversing the same soil constantly (approximately a circle with a 20 km radius), a very high resolution map of the base and surrounding area can be constructed and stored in the rover's memory. it can then be used to compliment its local imaging system so that even faster movement can be achieved later. Once the area is completely mapped by one or more of the rovers, the data can be transferred among the rovers so that each vehicle has the same topographical data base from which to draw .

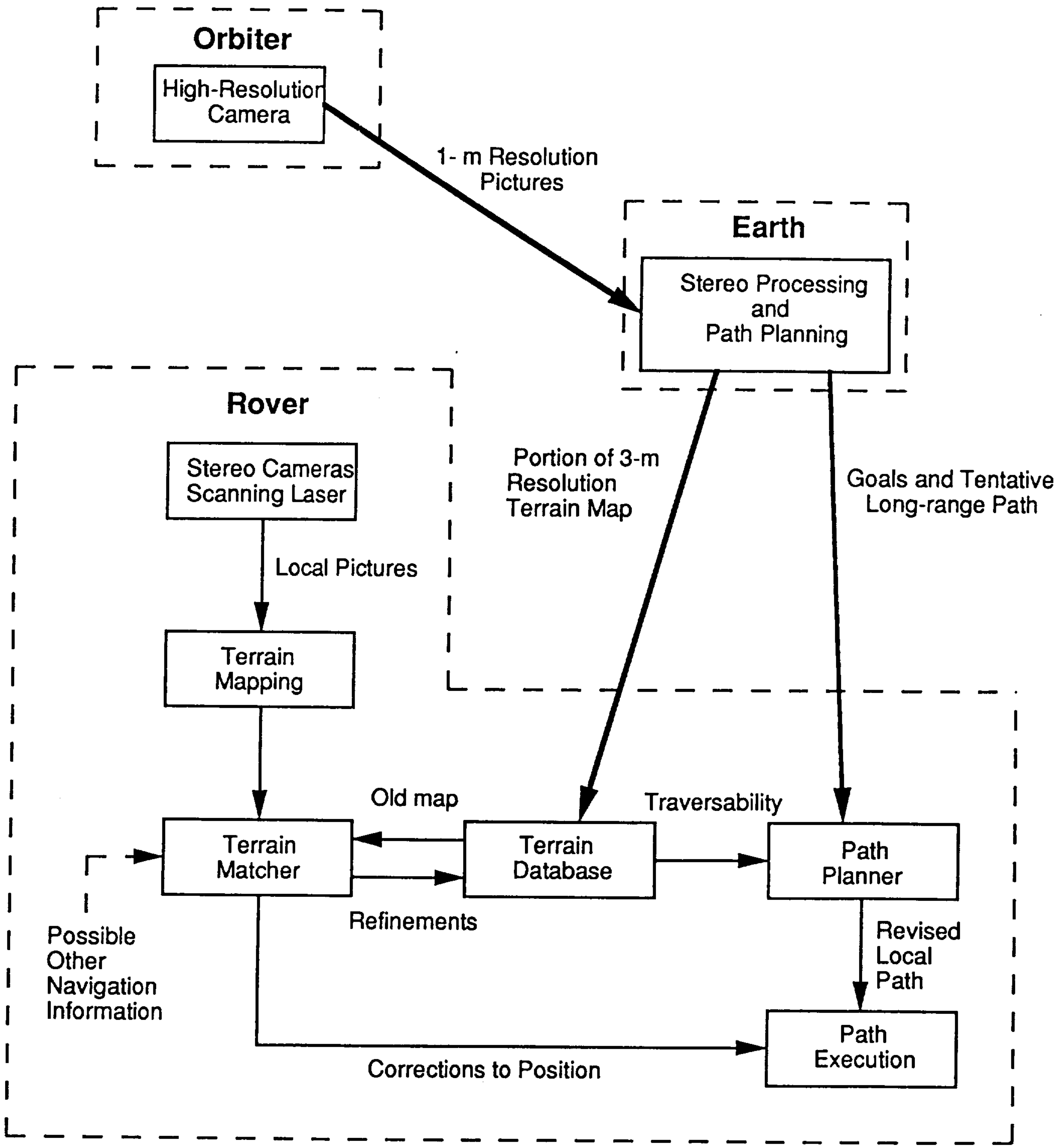


Figure courtesy of the Jet Propulsion Laboratory

Fig 10.4. Semi-Autonomous Operation of a Lunar Rover.

11.0 Management Structure

In order to accomplish the goals of this project, Lone Star Aerospace was divided into a management staff, seven technical departments, and a model team. The management staff consists of a project leader (the single point of contact between the contractor and the contractee), a systems integration leader, a chief technical officer, and an administrative officer. This staff oversees the work performed by the seven departments, assigns work, develops work schedules, and ensures that the design team meets all deadlines. Figure 11.1 shows the structure of the management staff and its members.

The seven departments consist of a trajectory team, a communications/power team, a spacecraft design team, an instrumen-tations team, a habitat/transportation team, a thermal subsystems team, and a robotic system team. Each of these departments had a department leader who was responsible for overseeing the work done in that department, making design decisions for that department, and reporting accomplishments and problems directly to the chief technical officer.

The design team consisted of a total of ten members. Each of these ten were assigned to at least two of the six initial departments (five original subsystem departments and the model group), so that each team had at least three members. The model design team consisted of three members who developed and designed a model and a poster characterizing the final design of the lunar base. The five technical teams completed their work involving in their technical specialties by mid-semester. After mid-semester, the ten members were re-assigned to one of two new technical groups, the thermal group and the robotics group. Thus, each subsystem had five members for its design team.

Under this organization, final integration of the design was handled by frequently holding group meetings, where all ten memebers of the design team gave their individual inputs. Integration was made more efficient by having a staff member whose sole job was ensuring that the integration of the various subsystems went smoothly. Figures 11.2 and 11.3 show the structure of the design team at the department level both before and after the mid-semester re-organization.

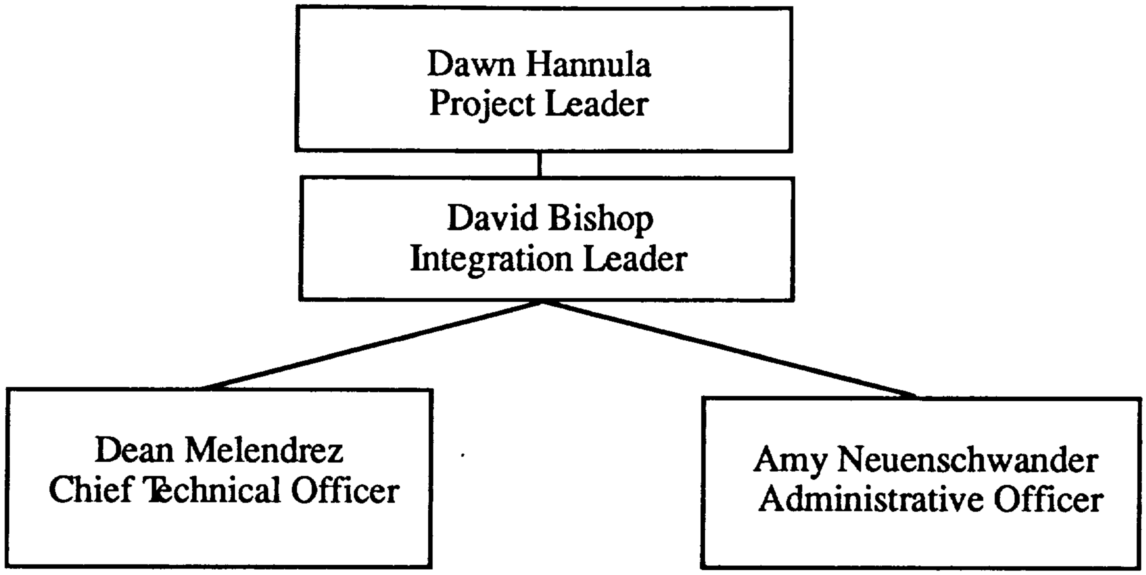


Figure 11.1. Management Staff and Structure.

It can be concluded from Figures 11.1, 11.2, and 11.3 that the organizational structure allowed for good communications as well as the rapid identification and correction of problems. This type of structure played a large role in the timely and successful completion of this project. The management staff of Lone Star Aerospace developed a schedule and a PERT/CPM chart for this design project. The timeline used to schedule and monitor the team's progress through each phase of the design is shown in Figure 11.4. It should be noted that the design team met all milestone dates agreed upon at the beginning of the project. The PERT/CPM chart, depicting how each phase of the program was controlled, is shown in Figure 11.5.

Trajectory

*Sanjiv Patel
Bill Hargus
Chris Niemann

Communications/Power

*Brett Padgett
Lee Wiesehuegel
Amy Neuenschwander

Spacecraft Design

*Lee Wiesehuegel
David Bishop
RudiChakrabarty

Habitat/Transportation

*Rudi Chakrabarty
David Bishop
Dean Melendraz

Instruments/Equipment

*Chris Niemann
Bill Hargus
Dawn Hannula
Sanjiv Patel

Model Design

*Bill Hargus
Brett Padgett
Dawn Hannula

*Denotes Department Leader

Figure 11.2. Primary Department Structure.

Thermal Systems

*Dean Melendrez
Dawn Hannula
Rudi Chakrabarty
Amy Neuenschwander
Chris Niemann

Robotics Systems

*David Bishop
Bill Hargus
Lee Wiesehuegel
Sanjiv Patel
Brett Padgett

Figure 11.3. Department Structure After Mid-Term.

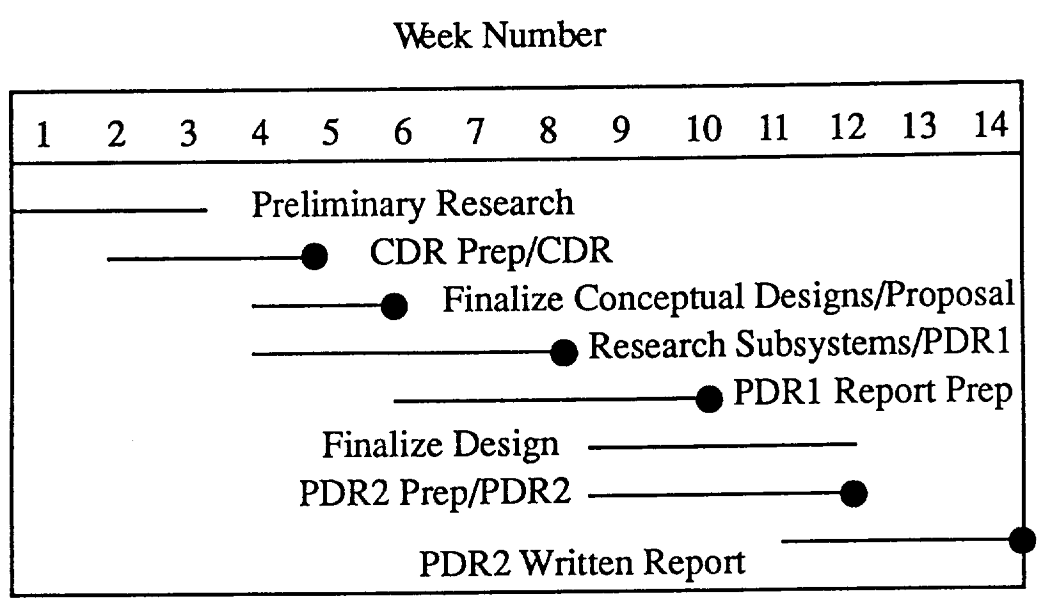


Figure 11.4. Project Timeline.

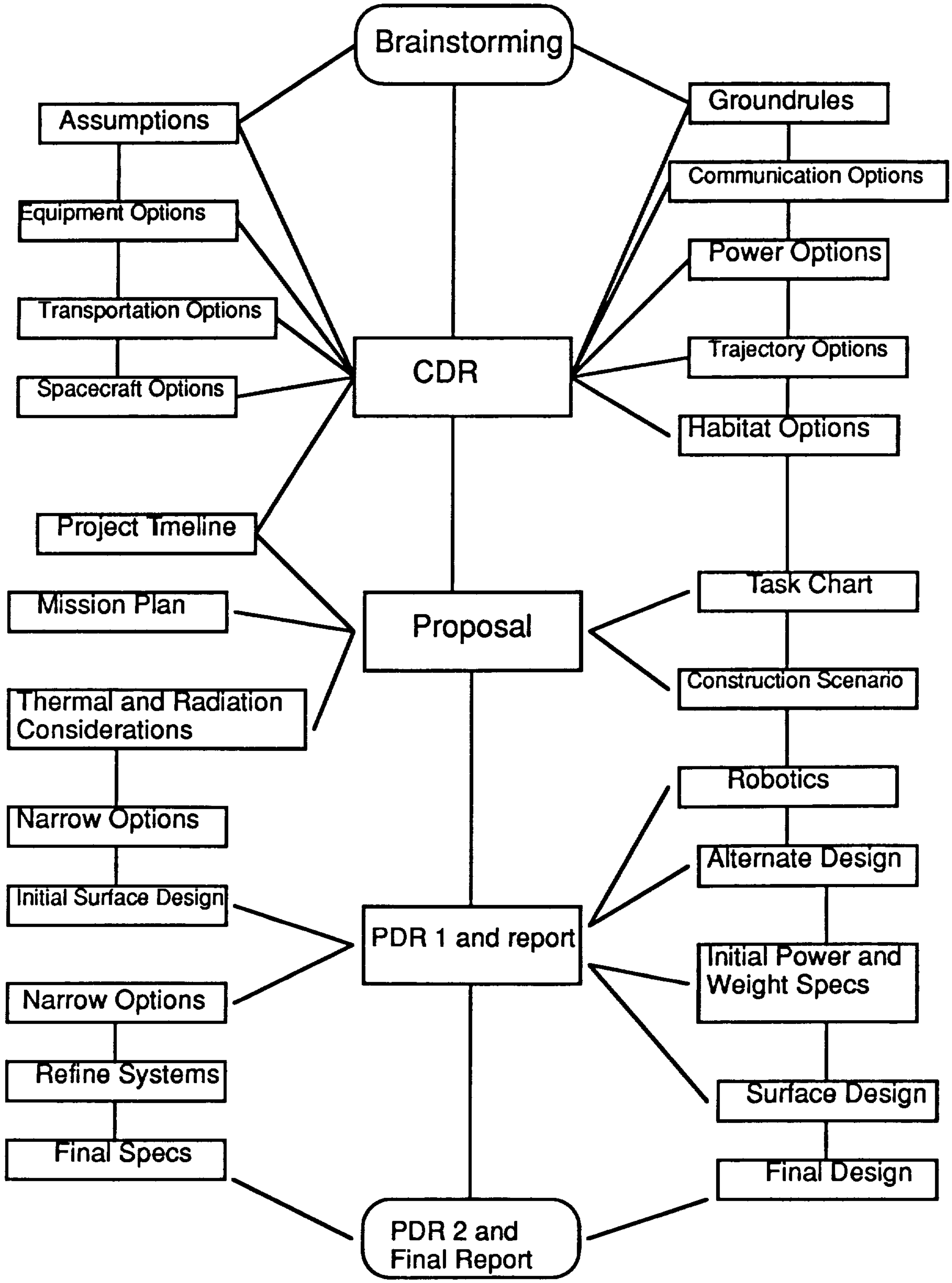


Figure 11.5. PERT/CPM Chart.

12.0 Design Cost

The following sections show the cost for completing this project. The individual cost for man-hours and hardware are presented separately. Total cost is then calculated. Finally, a discussion of performance and delivery is presented.

12.1 Cost of Man-hours

The cost per man-hour shown in both the proposal and the progress report are once again shown here. These figures were calculated based on the figures presented in the RFP (adjusted to include inflation since the printing of the RFP) [1] and from projected costs incurred by a previous design group constructing a similar base [2].

Original Formulation of Projected Personnel Cost

Weekly breakdown:

4 staff managers:	5 hours @ $35/hr each	$ 700.
10 engineers	10 hours @ $20/hr each	$ 2,000.
Total Weekly Personnel Cost:		$ 2,700.
Projected cost for 15 weeks:		$ 40,500.
Ten percent error estimate:		$ 4,050.
TOTAL PERSONNEL COST ESTIMATE:		**$ 44,550.**

The number of man-hours utilized over the past fifteen weeks has been recorded. This data was plotted against the estimated man-hour cost over that period and is shown in Figure 12.1. From this figure, the man-hour cost for this project was calculated to be $43,680.

Formulation of Fifteen Week Personnel Cost

Engineers:	10 @ $20./hour X 161 =	$ 32,200.
Management:	4 @$35./hour X 82 =	$ 11,480.
TOTAL MAN-HOUR COST AFTER FIFTEEN WEEKS:		**$43,680.**

12.2 Projected Cost of Material and Hardware

The material and hardware costs presented in the proposal were based on the expenses incurred by a previous design company in completing a similar project [2]. Computer costs consist of the cost of computer hardware, software, and mainframe time utilized. The original cost figures for materials and hardware have been reproduced below.

Original Formulation of Projected Materials and Hardware Cost

15 week rental of Macintosh computers:	$ 4,000.
15 week rental of IBM personnel computers:	$ 2,600.
10 VAX accounts @ $50 each:	$ 500.
Photocopies @ $0.10 each:	$ 65.
Laser printing @ $2.50 per page:	$ 825.
Transparencies @ $0.75 each	$ 50.
Model and poster materials:	$ 100.
Subtotal:	$ 8,140.
Ten percent error estimate:	$ 814.
TOTAL MATERIALS AND HARDWARE COST ESTIMATE:	**$8,954.**

The cost of materials and computer time has been recorded over the past fifteen weeks. This information is shown below. All amounts were rounded up to the nearest dollar.

Formulation of Material and Hardware Costs after 10 Weeks

15 week rental of Macintosh computers:	$ 4,000.
15 week rental of IBM personnel computers:	$ 2,600.
15 VAX accounts @ $50 each:	$ 750.

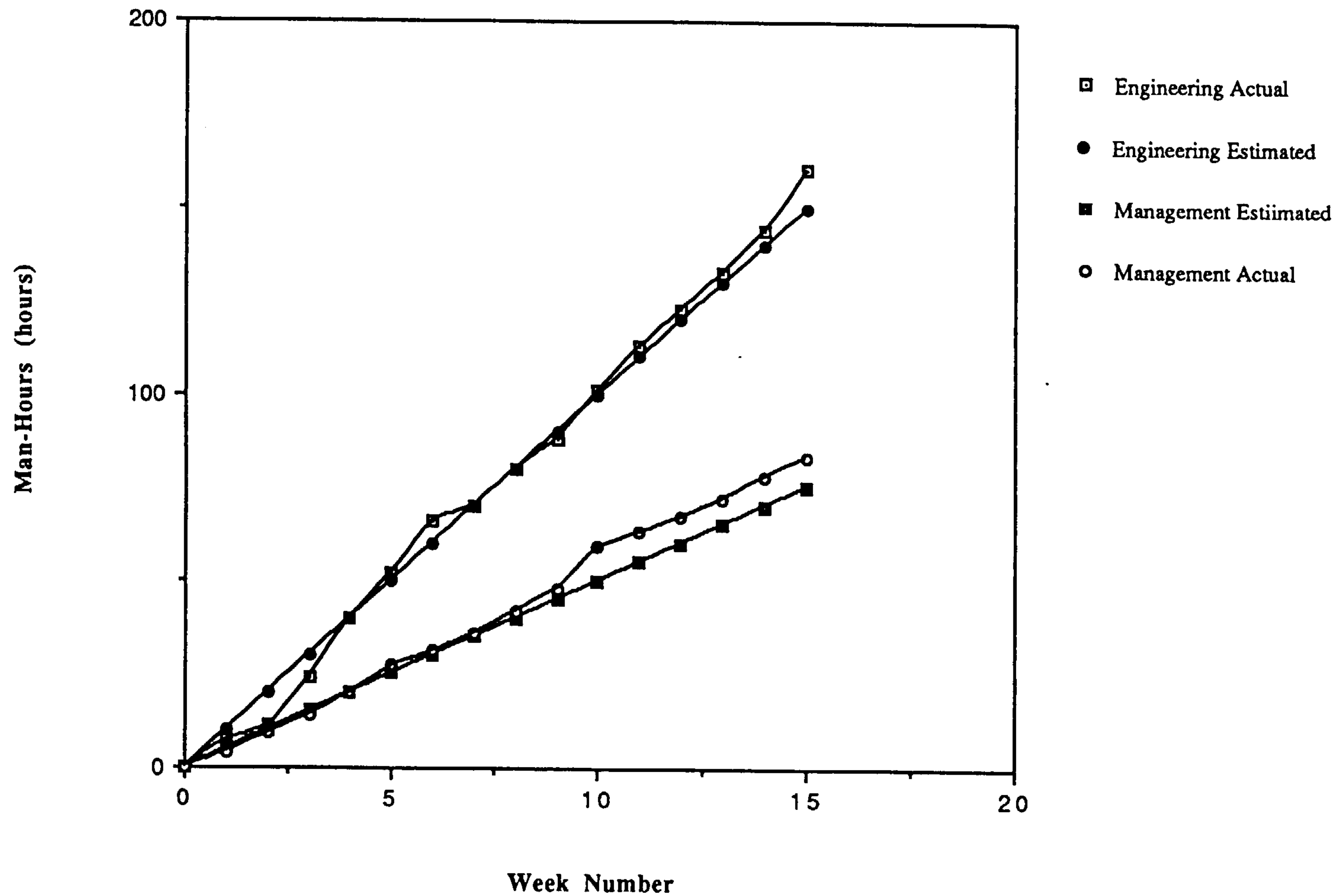


Figure 12.1. Man-Hours vs. Weeks.

Photocopies: 1,250 @ $0.10 each:	$ 125.
Laser printing: 230 @ $2.50 per page:	$ 575.
Transparencies: 130 @ $0.75 each	$ 98.
Model and poster materials:	$ 25.
TOTAL MATERIALS AND HARDWARE COST AFTER 15 WEEKS:	**$8,173.**

In comparing this figure to the estimated figure given at the beginning of this project, it can be concluded that this project is under budget in the area of material and hardware cost.

12.3 Projected Total Project Cost

The projected total cost for the entire project was calculated in the proposal. It was estimated that the project cost would total $65,054. However, in examining the 15 week figures, it was found that the total cost for this project was $51,853.

Formulation of Total Project Cost

Total personnel cost:	$ 43,680.
Total materials and hardware cost estimate:	$ 8,173.
TOTAL PROJECT COST:	**$51,853.**

12.4 Performance and Delivery

It was stated in the proposal that if Lone Star Aerospace fails to deliver any one of the operating systems or is more than 10 days late in delivering an operating system, the company agrees to provide a 15% deduction to the total personnel costs to the contractee (payable upon final delivery of the recommended design). However, Lone Star Aerospace met all milestones and delivery dates. Therefore, this deduction does not apply.

12.5 List of Deliverables

The following is a list of deliverables which Lone Star Aerospace produced as well as the dates on which they were ready.

Preliminary Design - Stage 1 Briefing	16 October 1991
Preliminary Design - Stage 1 Report	30 October 1991
Preliminary Design - Stage 2 Briefing	30 November 1991
Final Report	06 December 1991

13.0 Legal Analysis

Additionally, a legal analysis of the FLARE mission has been performed. The effect of environmental considerations with respect to this mission were taken into account in the generation of this analysis. The findings of this analysis can be found in Appendix G.

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Appendix A: Instrumentation Package Details

The following is a brief description of the various astronomical instrument packages to be placed on the lunar surface. Because of its unique suitability to the far side of the moon, particular attention will be paid to the VLF array.

A.1 Very Low Frequency Array (VLFA)

The Very Low Frequency (VLF) range is defined to be between 1 and 30 MHz. The low end of this frequency range is useful in examination of the interstellar medium since the large scale distributions are observable at these frequencies. Additionally, galactic and extragalactic sources are observable at the higher end of the VLF range. This range of frequencies is ideally suited for studies of phenomena not observable in any other spectral band. These phenomena include low energy cosmic rays particles, thermal environments of discrete radio sources, and coherent radiation arising from collective plasma sources. [1]

Very low frequency astronomy is presently limited by two important constraints. The first is the ionosphere of the Earth. It has a characteristic variable plasma frequency of approximately 10 MHz. This combined with both manmade and geomagnetic radio interference limit routine observations to frequencies greater than 30 MHz. The second and more limiting constraint is the Interstellar Medium. It absorbs and surpresses radio emissions in various ways. This sets fundamental limits on the achievable resolution of direct VLF observations. These limits are approximately 0.5° at 1 MHz and 2" are 30 MHz [1].

At present, VLF astronomy is extremely limited. Observations must be undertaken above the ionosphere. Several direct satellite observations have been made, but these experiments have been few and with rudimentary equipment [1]. The lunar farside will be an excellent location for a Very Low Frequency Array (VLFA). This is due to the fact that the mass of the moon will block the VLF radio waves emanating from the Earth. The lack of a significant lunar atmosphere also makes the far side lunar surface an ideal location [2].

The Very Low Frequency Array (VLFA) will consist of 300 one meter dipoles in a power law distribution over an 18 km diameter circle. A central station will provide the communications link and preliminary processing of the instrument data. The operating frequencies will range from 1 to 30 MHz. The mode of the VLFA will be as an interferometer rather than a phased array [3].

The VLFA will consist of 300 dipoles units. As seen in Figure 1, the dipole units will consist of two 1 meter dipole antennas, a box to house the electronic, and an antenna to relay data back to the main station [4]. Each of the 300 dipoles will require wires one meter in length, solar cells and batteries for power, a receiver and transmitter requiring radiation-hardened chips, and thermal insulation so that the temperature bounds of the instruments will not be exceeded. The mass of each dipole will be between two and five kilograms. The power required will be a maximum of 1 We.

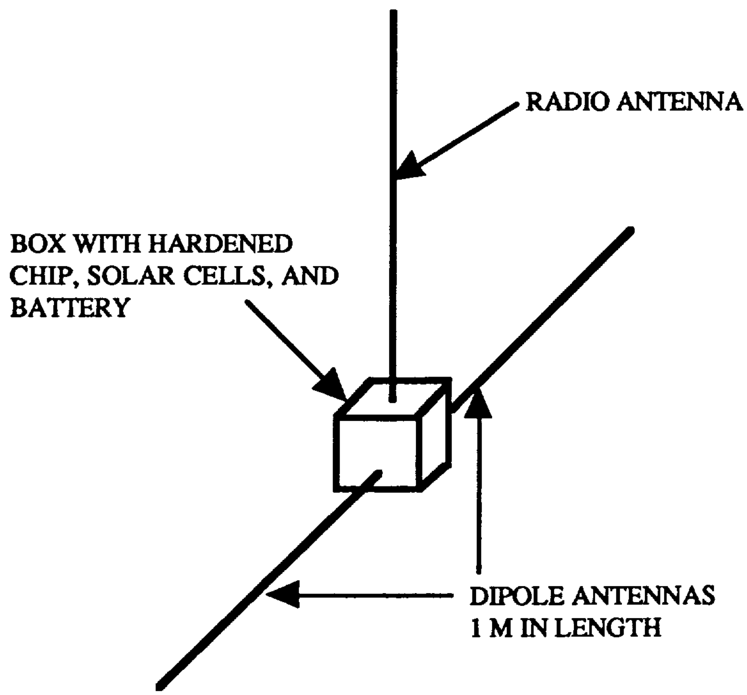

Figure A.1. Dipole Unit [4].

However, it has been suggested that the power required may be only in the milliwatt range. Small photovoltaic arrays will be able to supply the required power for each dipole. During the long lunar night, power will be supplyed by long life batteries charged by the photovoltaic cells during the lunar day [3]. The chosen communications link will be high frequency radio. The advantage of high frequency radio is that the chip at each dipole unit will act as a phase-stable radio relay, thus simplifying the dipole units [4].

In addition to the dipoles, there will be a central station that will sort and process the data from the dipole array. It will house computers and communications equipment for the instrument as a whole. For these reasons, the main station must be shielded from radiation due to cosmic rays and solar flares. The shielding material of choice is lunar regolith with a thickness of two meters. Heat pipes will be used for waste heat rejection. The mass of the central station will be approximately 800 kg. The maximum anticipated power requirement for the central station will be 500 to 1 kWe [3]. The required power will be provided by a series of Radioisotope Thermal Generators (RTGs) due to the large size and distance the array will be placed from the main FLARE observatory complex. Communications to the FLARE observatory complex will be by fiber optic cable.

The total mass of the entire VLFA will be a maximum of 2,300 to 1,400 kg. The VLFA will be the first instrument to examine the VLF range. The total VLFA is expected to have a power requirement of 100 We and to require a data transmission rate of 32 kbps [5].

A.2. Moon-Earth Radio Interferometer (MERI)

The far-end link of the Moon-Earth Radio Interferometer (MERI) will be a radio antenna, 15 m in diameter, placed on the Moon. Placement of the antenna on the lunar far side will offer an interferometer with a 384,400 km baseline when readings from the lunar dish are correlated with readings from similar dishes on the Earth (see Figure A.2). For future expansion, other radio antennae can be sent into orbits between the Earth and Moon in order to obtain even better sensitivity and mapping capability for the MERI. The system will enable better astronomical and synthesis mapping measurements to be made, and offers the possibility of improving the known celestial coordinate system. The total system is expected to weigh 2.1 metric tons, with a power requirement of 15 kWe and a data transmission requirement of 10,000 kbps [5]. The initial system is expected to have a resolution of 0.4 micro-arcseconds at 300 GHz.[3]

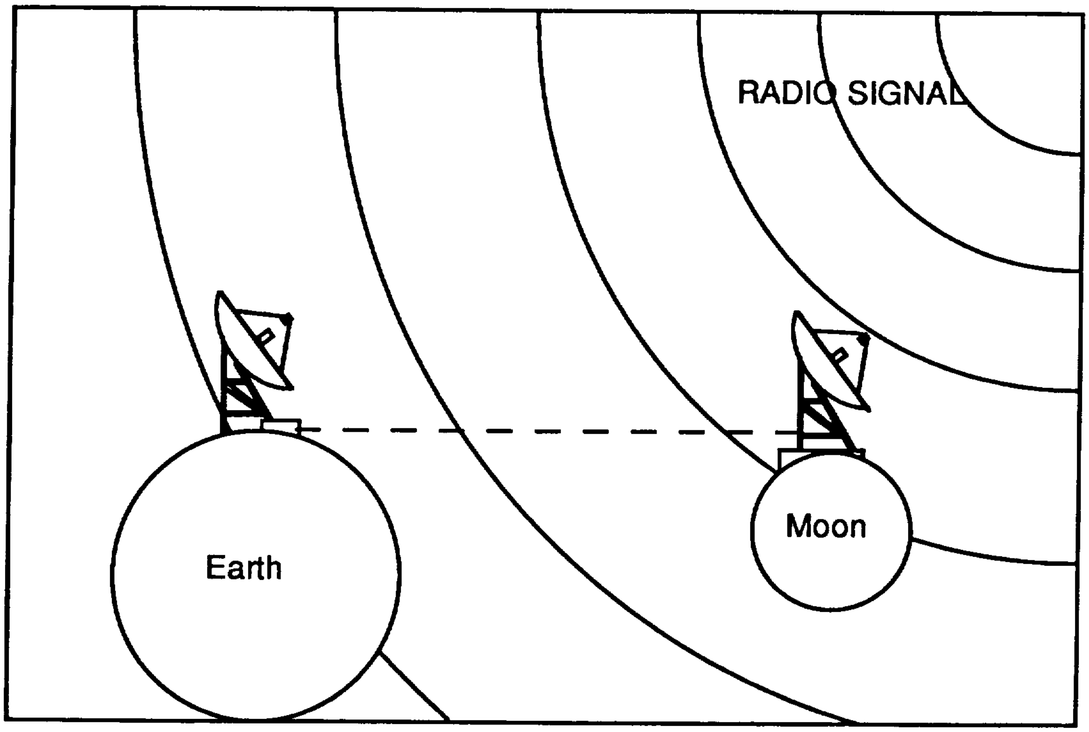

Figure A.2. Correlation of MERI Radio Antennae.

A.3 Lunar Transit Telescope (LTT)

The Lunar Transit Telescope (LTT) will take advantage of the long integration times provided by the slow lunar rotation rate to perform an in depth survey of the sky. Although the 1.5 m telescope is fixed, the Charge-Coupled Diode (CCD) sensor array is programmed to shift its register to compensate for the lunar rotation rate. This technique allows the superposition of successive views of the same sky area, making the telescope extremely sensitive to even faint emissions. The LTT will investigate the infrared, ultraviolet, and visible portions of the spectrum while scanning a 2° wide strip of the lunar sky. The telescope will use five, broad spectral bands in its imaging, looking at wavelengths from 0.1 to 2 μm, and will have a spatial resolution of 0.1 arc seconds [6]. Because it uses CCD sensors, the LTT will need to be actively cooled to 100 K by cryogenics as well as using some kind of passive sunshade. The telescope will weigh 1.3 metric tons, have a data transmission rate of 30 Mbps, and will have a power requirement of about 400 We [5].

A.4 Optical Interferometer (OI)

The Optical Interferometer (OI), when fully constructed, will enable the FLARE project to observe the parallax of objects to several megaparsecs and to image accretion disks around massive black holes. The OI, as shown in Figure A.3, will consist of twelve 1.5 m optical telescopes which are fixed on the lunar surface in a Y-shaped array. Each of these elements should weigh about 1.3 metric tons apiece. Also included in the final array are a central correlating station and twelve movable optical delay line carts on tracks. These carts will provide coarse optical path length adjustment by carrying mirrors along straight line paths. Each arm of the OI will be six kilolmeters long. This will give the OI a maximum baseline of 10 km for the instrument and a spatial resolution of 1 micro-arcsecond [6]. Each telescope will have its optics passively cooled, possibly with the use of sunshades, while the telescope sensors will have to be actively cooled to less than 100 K by cryogenics. The positioning of the elements of the OI will have to be controlled very accurately. In fact, the elements will have to be positioned to within 10 nanometers over a distance of close to 20 km [7]. However, the lunar surface offers an exceptionally stable base for the instrument, and laser techniques seem adequate for positioning of this accuracy. The total mass for the OI should be 16 metric tons, with a power requirement of 9 kWe and a data rate of 1 Mbps [5].

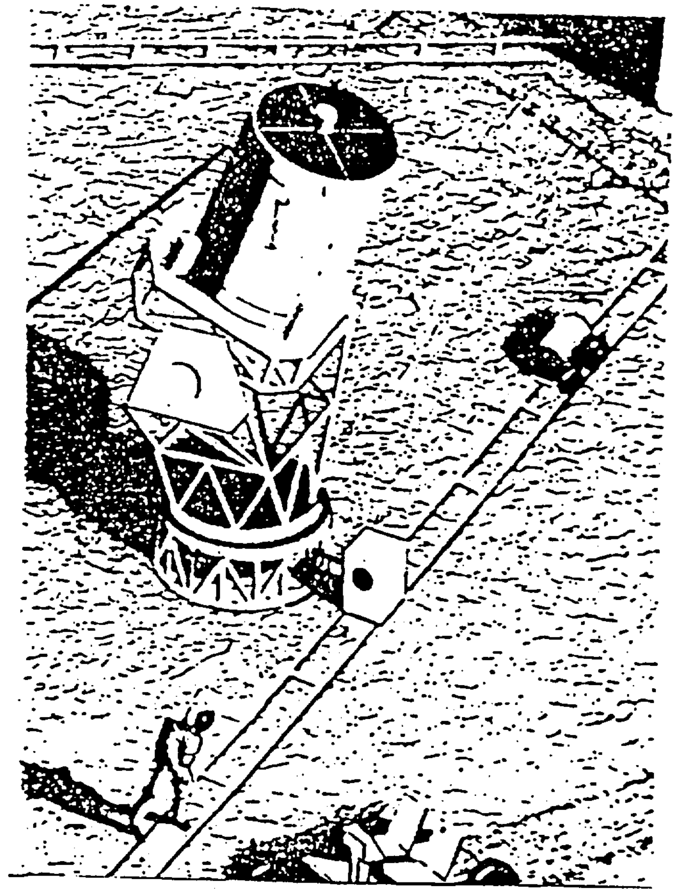

Figure A.3. Optical Interferometer Element and Y Shaped Array [8].

A.5 Submillimeter Interferometer (SI)

The Submillimeter Interferometer (SI) will provide the ability to resolve protostellar disks in regions where stars are formed and to study star burst phenomena in distant galaxies [6]. The SI will eventually consist of seven 5 m diameter elements arranged in a Y-shaped array similar to the OI. Each of the arms will be 6 km long, giving the instrument an effective baseline of 10 km. When fully set up, the SI will be able to survey the band of the spectrum from 30 μm to 1 mm with an angular resolution of 1 to 10 milli-arcseconds [6]. The antenna units will need to be passively cooled by sun shields and will also use detectors actively cooled down to 100 K. Each antenna will weigh two metric tons, which gives a total package mass of 14 metric tons. The power requirement will be 20 kWe and the data rate will be 100 kbps [5].

A.6 16m Optical Telescope

The 16m Optical Telescope will consist of four 4 m optical telescopes interlinked with one another to give an effective aperture of 16 m, and will offer a major scientific step in optical imaging over the HST. With the ability of this telescope to conduct spectroscopic observations of faint objects, it should be possible to both detect Earth-like planets circling other stars and to study star formation [6]. Also, the telescope may be combined with the Optical Interferometer to produce an even more effective instrument. It is expected that the telescope will have a field of view of about 1 arcsecond, and will be able to track an object to about 1 milli-arcsecond [6]. The four meter telescopes will be constructed from hexagonal elements, as shown in Figure A.4, and will probably use sun shields to passively cool the structure. Additionally, the CCD detectors will need to be cooled actively to about 100 K. It is estimated that the initial four meter telescope plus the correlation package will weigh 15 metric tons, with the other four meter segments each weighing about nine metric tons apiece. The power requirement should be about five kWe and the data rate should be about 10 Mbps [5].

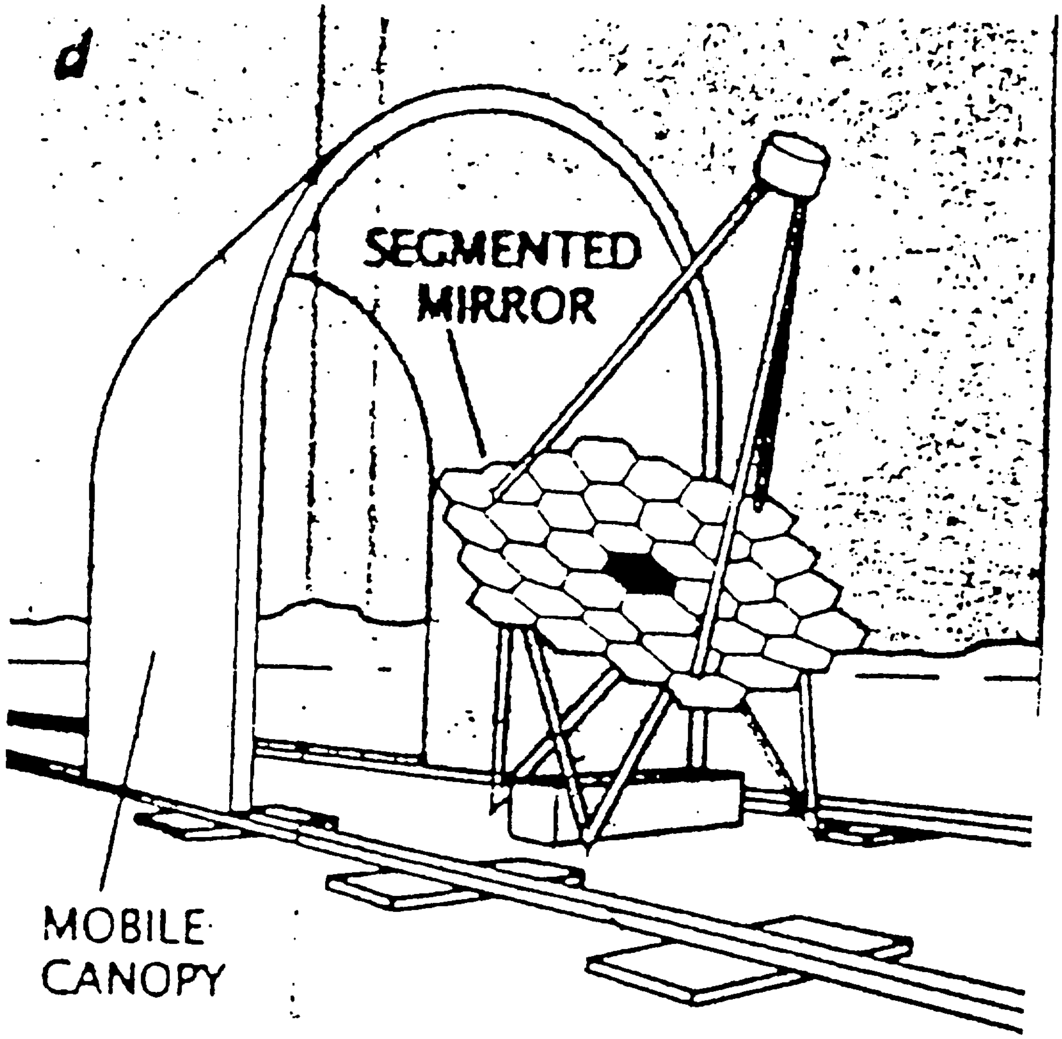

Figure A.4. 4 Meter Telescope with Hexagonal Segments [8].

A.7 High Energy Cosmic Ray Detector (HECRD)

The High Energy Cosmic Ray Detector (HECRD) will help to sample directly accessible matter from outside the solar system by looking at particles in an energy range from 1,000 GeV to 10 million GeV. Because it is difficult to differentiate between particles at these high energies, stationary nuclei must be provided as targets with which the incoming cosmic ray particles can collide. The HECRD is well suited to the lunar surface, since the lunar regolith can be used as target mass in the detectors. Indeed, the detector will consist of layers of regolith target mass alternating with plastic scintillators or drift chambers through a depth of 3.6 meters. The mass shipped to the Moon is expected to be three metric tons, with a power requirement of one kWe and a data rate of 10 kbps [5].

A.8 Low Energy Cosmic Ray Detector

The Low Energy Cosmic Ray Detector will allow scientists to explore the physical processes responsible for forming elements by sampling galactic cosmic radiation. This radiation is made up of atomic nuclei which have been accelerated to near light speed. These nuclei thus carry matter from supernovas and other galactic bodies [5]. The Moon lacks sufficient atmosphere or magnetic field to shield the surface from the radiation, thus making it a good site for cosmic ray detectors. The detector itself will be a cylindrical ion chamber containing argon gas. The dimensions of the chamber are envisioned as being 2.5 m in diameter and four meters in height. The total mass should be about three metric tons, with a power requirement of 500 We and a data rate of 10 kbps [5].

A.9 High Throughput X-ray Telescope

The High Throughput X-ray Telescope will be used to study X-ray background radiation in the energy region from 10 to 50 keV. The telescope will be a large, unpointed transit telescope with a collecting area of about 5×10^5 cm^2 [8]. The telescope mirror will consist of flat, grazing-incidence reflector plates. The main telescope package should resemble the LAMAR X-Ray telescope shown in Figure A.5. An imaging hard X-ray detector is placed separately, and will have a spatial resolution of one millimeter. The total mass of the X-ray instruments is expected to be two metric tons, with a power requirement of one kWe and a data rate of 100 kbps [5].

Although the High-Throughput telescope is the choice for the X-Ray portion of the FLARE project, it should be noted that other X-Ray packages are possible. In particular, later expansions of the FLARE base could include a broad sky coverage X-ray telescope with a wide field of view. This instrument could resolve thousands of of objects simultaneously and study the temporal behavior of the X-Ray sky. A picket-fence collimator set-up, which could yield a telescope with an extremely long focal length, is also a potential candidate for expansion plans [8]. However, such an instrument would be beyond the scope of the FLARE projects construction plans.

A.10 Gamma-Ray Telescope (GRT)

The Gamma-Ray Telescope (GRT) will provide the ability to look at enigmatic bursts of gamma-ray energy that rise hundreds of times above the normal gamma-ray background. The telescope will also monitor other transient phenomena such as solar flares, supernovas, and flare stars. The GRT is envisioned as an array of germanium detectors placed behind a coded mask, which will monitor an energy range from 10 keV to 10 MeV [4]. The array will be placed in a six meters deep excavation in the lunar surface and shielded by a two meter thick ceiling of lunar regolith. The GRT is expected to weigh three metric tons, with a power requirement of one kWe and a data rate of 25 kbps [5].

A.11 Environmental and Magnetospheric Surveys

The environmental and magnetospheric surveys will be conducted with the use of relatively small instrument packages. The environmental package will consist of several instruments to measure basic data such as radiation, temperature, seismic activity, and atmospheric composition. This package can also be used to monitor the effects of the construction of the lunar base on the lunar environment. It will have a power requirement of approximately 10 We, a data transmission rate of one kbps, and a mass of approximately 100 kg [5].

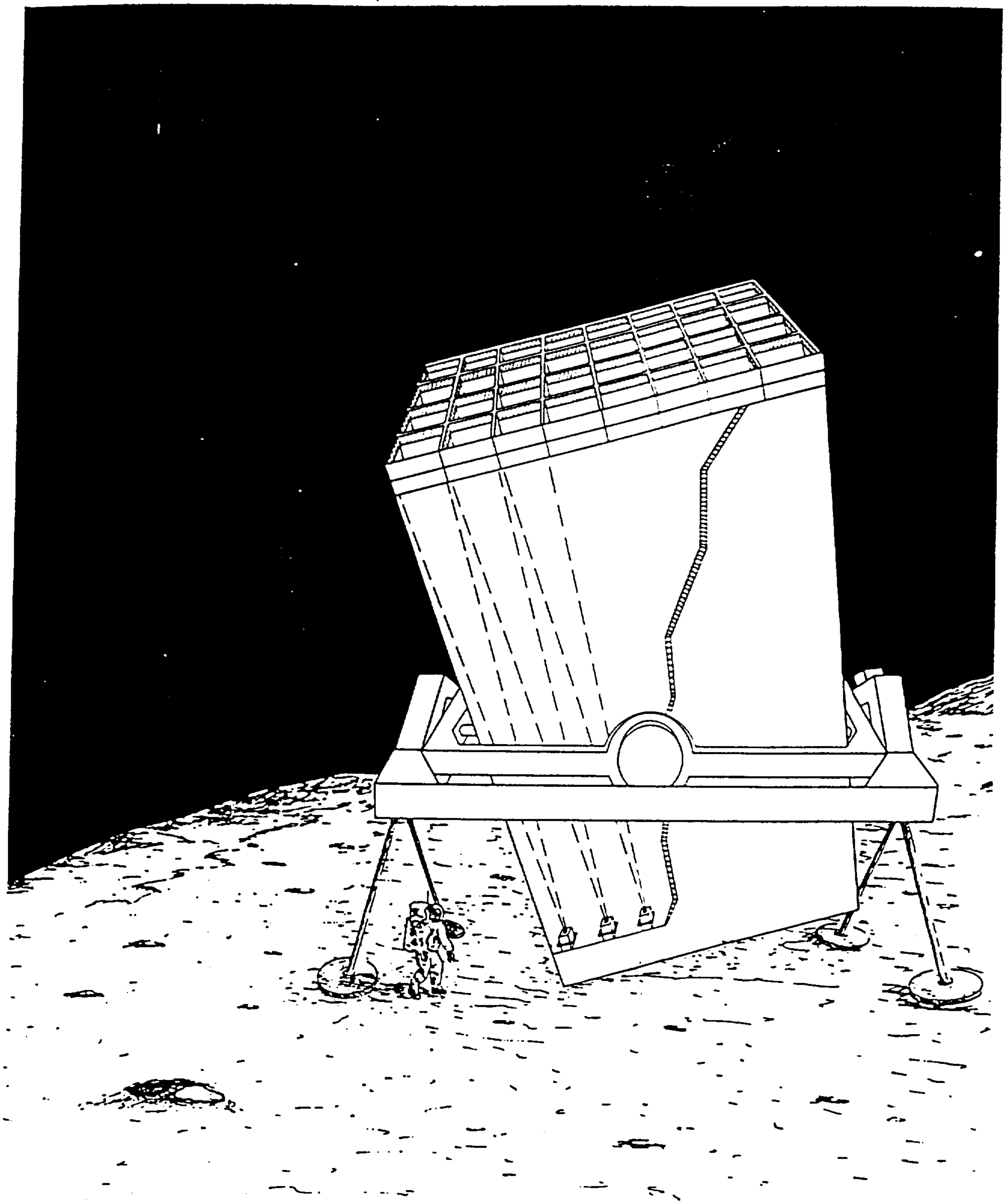

Figure A.5 LAMAR X-Ray Telescope [8].

The magnetospheric package will monitor the interaction of the solar wind with the Earth's magnetic field. The package will consist of a magnetometer, solar wind detector, and photon and atom imaging instruments. It will have a power requirement of 30 We and a data transmission rate of one kbps [5]. With a mass of only 10 kg, the magnetospheric package can be easily deployed by an astronaut.

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Appendix B: Geoscience Survey Details

Although the main focus of FLARE mission will be the establishment of a far side lunar astronomical observatory, lunar geoscience will also be an area of important investigations. The six successful Apollo missions returned 382 kg of samples from the lunar surface, while the three Soviet Luna unmanned landers returned 250 kg. From these samples and various photographs, geologists have advanced many theories and speculations about the origin and age of the Moon, the absence of a lunar magnetic field, the composition of the lunar mantle, and other items of importance to lunar geoscience [1, 2]. The construction of the far side observatory is an excellent opportunity for a limited amount of exploration and discovery for the advancemant of lunar geoscience.

B.1 Lunar Geoscience Background

The FLARE observatory will be located in the far side crater of Hertzsprung. Hertzsprung is a multi-ring basin with distinct inner and outer rings. These rings have diameters of 260 km and 570 km, respectively [1, 3]. The mechanics of the formation of multi-ring basins are not completely understood and have never been adequately explained. However, there is no disagreement that the large impacts required to produce multi-ring basins are of extreme interest to lunar geoscience [1].

Most of the far side of the Moon, including Hertzsprung, is of the terrain known as the lunar highlands. The lunar highlands are characterized by the oldest terrain on the Moon. They are saturated with ringed basins and large craters [1]. The entire lunar surface, including the lunar highlands, is covered with a surface layer known as regolith. The lunar regolith is the continuous surface layer of the Moon which is generally several meters thick [4]. More specifically, the lunar regolith is a layer of poorly sorted, relatively fine, rocky debris. It contains ejected material from distant cometary and meteoric strikes. Approximately fifty percent of the regolith is rock and mineral fragments, while most of the remaining fifty percent consists of glass fragments [5].

B.2 Geoscience Instruments and Investigations

The following sections discuss the geological instruments that will be included in the packages to be sent to the lunar surface.

B.2.1 Seismometers

Seismometers will be used to measure extremely small vibrations on, or near, the surface of the Moon. As shown in Figure B.1, the instrument will be small, compact, and capable of being placed on the surface with a protective heat shroud. These instruments will be advanced versions of those seismometers placed on the lunar surface in each of the Apollo landings [6]. These instruments will need to be updated to have a ground sensitivity of 0.3 to 0.03 nm (0.03 nm being preferred). The frequency band width should also extend between approximately 30 and 0.03 Hz. In addition, the placement of the seismometers at least one meter below the surface of the regolith will be needed to limit thermal variations. Ideally, these seismometers should be placed at locations across the lunar surface [2]. They will passively detect Moonquakes and meteor strikes, provide information of the nature of the interior of the Moon, and detect the number and size of meteorites that strike the lunar surface [6].

B.2.2 Geophone Array

Surface geophone arrays will be used in active seismology tests to determine seismic data on large-scale structures on the Moon. This will be done by detonating explosives on the lunar surface. The time that the sound waves take to transmit through the regolith and reach individual instruments on the array will be precisely measured. Sound wave reflections from these tests off subsurface structures will provide further data as to what may lie below the lunar regolith [6].

Particular interest will be placed in studying the nature of the rings of multi-ring basins such as Hertzsprung. Therefore, the geophone will be placed in this location. The geophone will consist of a 10 instrument array with each element closely placed in a line (with distances of 100 m between each element) on the multi-ring crater. This geophone will be portable and reusable [6]. In addition, it is not anticipated that the detonations will have a significant effect on the other instruments if a safe distance from astronomical instruments is maintained. A safe distance will be assumed to be 15 km until experience is gained and the safe distance may be redefined.

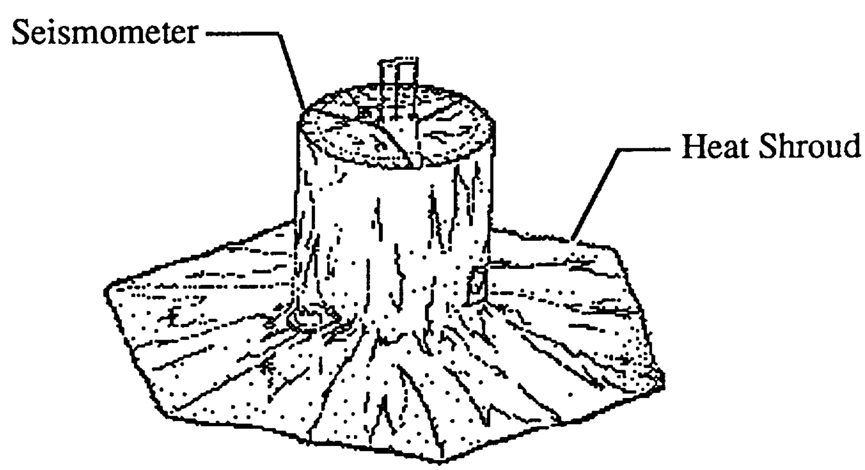

Figure B.1. Lunar Seismometer.

B.2.3 Heat Flow Measurement

The fundamental laws of thermodynamics state that heat flows from hot regions to cold regions. The Moon is known to have a warm core from heat flow measurements performed during the Apollo program, and similar measurements on the lunar far side would complement the data already obtained. Heat flow measurement would allow the rate of heat loss from the interior of the Moon to be determined. This is important since the heat now reaching the lunar surface is thought to be produced by radioactive decay. The heat flow measurement will provide data that may determine the levels of radioactivity now present inside the Moon [6].

Figure B.2 shows the configuration of the heat flow experiment from the Apollo missions. The FLARE heat flow measurement instrument will be physically identical but will incorporate more precise modern instrumentation into the design. The heat flow instrument will consist of two 3 meter holes drilled into the regolith robotically with a special drill attachment. Platinum resistance thermometers will be placed into each hole at several points at the lower portions of the holes. Less accurate thermocouples will be placed in the upper portions. Also, a probe to determine the thermal properties of the lunar material will be placed into each hole. The temperature of the regolith surrounding the hole will be thermally disturbed for several months. Temperature measurements will be taken over time and should approach the undisturbed temperatures of the Moon [6].

B.2.4 Drive Tubes

The drive tubes shown in Figure B.3 will be used to collect samples of the lunar regolith as a function of depth. The 45 cm thin-walled tubes may be fastened together in groups as large as four. They will be driven either robotically, or by visiting humans, into the lunar regolith to collect their samples. These samples will then be stored in the drive tubes until examination at a later time [6].

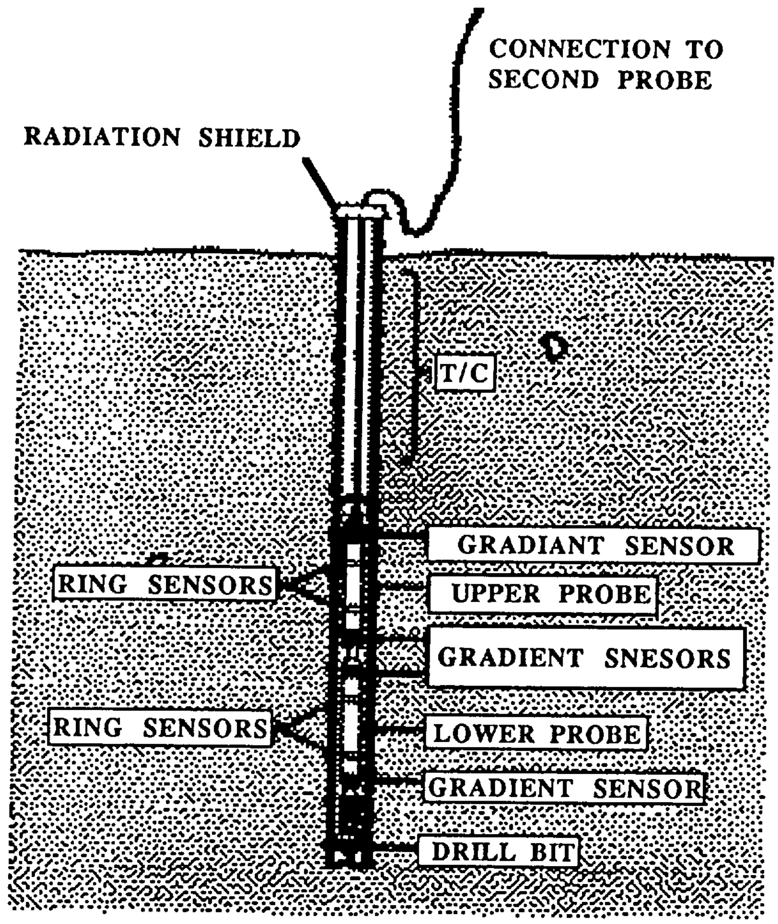

Figure B.2. Heat Transfer Measurement.

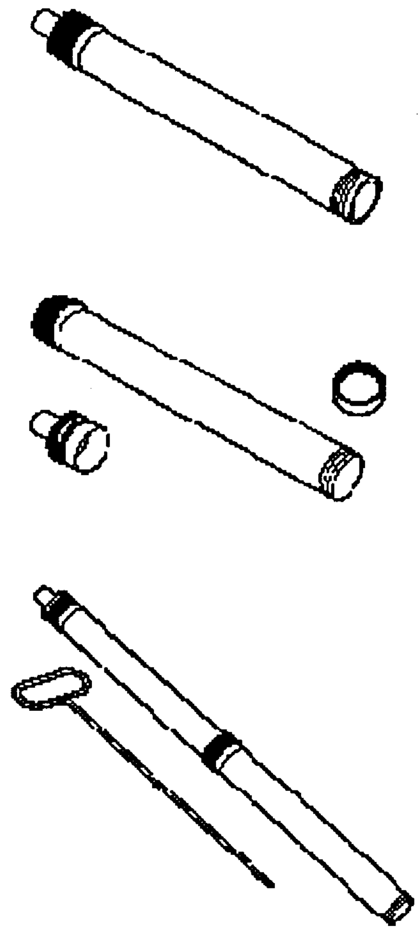

Figure B.3. Drive Tubes.

B.2.5 Humans and Robots

Humans will be the most versatile equipment used to study lunar geology. The Apollo missions recognized this. Each crew was trained to observe, orally record, and photograph the lunar geology during the lunar traverses. However, only one of the twelve astronauts placed on the surface of the Moon was actually a trained geologist [6, 7].

During the course of the manned missions to the FLARE observatory, excursions data (accompanied by the descriptions made by the astronauts) and photographs will be invaluable in the study of the geology of the Hertzsprung area. It is recommended that at least one of the crew members sent during the scheduled missions to the FLARE observatory be a trained geologist. The presence of trained science professionals will provide valuable insight if placed in situations that non-scientists crew members might not recognize.

Since the maximum allowable human stay on the FLARE base will be limited, any human geologists will be on the lunar surface for only short periods of time. Geologists may not always be available for selection as crew members. For these reasons, it is recommended that a number of small semi-autonomous roving robots with video cameras and manipulator arms be sent to the Moon as well. These roving robots will be based on a design currently being tested by the Jet Propulsions Laboratories (JPL) [8]. Currently, it is planned that three 25-35 kg rovers will be deployed in addition to the one sent in the initial lander of the precursor missions.

B.2.6 Experimental Setup

The setup of the geological instruments will be in a station arrangement. At each station there will be a set of heat flow measurement instruments and a seismograph. Each station will be monitored by a controller, which will consist of hardened electronics and a high gain antenna. Communications will be established with the FLARE observatory directly via fiber optic cable. The stations will be powered by a small Radioisotope Thermoelectric Generator (RTG) and will have a power requirement of less than 70 We. The stations are conceived to be limited versions of the Apollo geological testing stations [6].

In order to set up these stations, several sets of equipment are required. First, it will be assumed that the stations will be robotically placed by a general lunar vehicle from the FLARE observatory. Several modular equipment packages will be placed on this vehicle to install each station. Figure B.4 shows the drill used by the Apollo astronauts to drill each hole for the Apollo heat flow experiment. A similar configuration will be designed to attach to the general lunar vehicle and drill the holes as required for the heat flow measurement instrument [6].

The geophone arrays will not be placed in permanent stations. Rather, they will be robotically placed at various locations near explosive charges that will be remotely detonated. After the tests have been performed on a particular area, the arrays will be collected and used at subsequent sites. The lunar vehicle will also be used the complete this task.

The other instruments will be used as necessary. Core samples collected by use of drill tubes may either be robotically or manually performed.

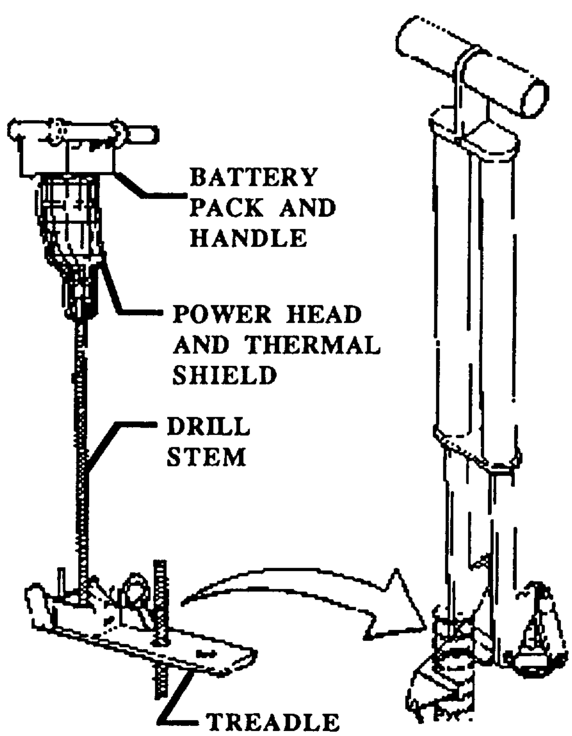

Figure B.4. Lunar Surface Drill.

B.2.7 Component Masses

Approximated masses for the outlined geological instrumentation are shown in Table B.1. It is believed that the maximum total mass of 1,305 kg is not excessive and represents a good investment for lunar geoscience.

Table B.1. Component Mass - Geological Instruments.

Unit Mass (kg)	Quantity	Net Mass (kg)
100	6	600
100	3	300
25-35	3	105
N/A	N/A	300
Total Mass		**1305 kg**

References

1. Taylor, Stuart Ross. *Planetary Science: A Lunar Perspective.* Houston: Lunar and Planetary Institute, 1982.
2. *NASA-LRC Lunar Orbiter Project-Mission V.* "Spacecraft Frame No. 024, 2 of 3 High Resolution." Photograph. Corp of Engineers, Aug. 8, 1967.
3. Powell, Corey S. "The New Moon." *Scientific American,* Aug. 1991.

Appendix C: Materials Processing Details

The main focus of FLARE will be the establishment of an astronomical observatory on the lunar far side. However, there is a secondary focus to FLARE. This secondary focus is to gain engineering experience on the lunar surface as a prelude to further exploration of the Moon and Solar System. An important area in which to gain engineering experience will be materials processing. A great deal of engineering knowledge could be gained by studying scale models of materials processing plants. FLARE has scheduled the construction of three scale-model materials processing plants. Oxygen will be extracted from the lunar regolith by vapor phase pyrolysis. Various structural metals will be produced by the hydrofluoric acid leach process (HF Acid Leach). In addition, magnetic extraction of meteoric iron refined by the gaseous carbonyl process will be examined.

C.1 Lunar Oxygen by Vapor Phase Pyrolysis

The vapor phase pyrolysis method of lunar oxygen production was chosen for several reasons. First, vapor phase pyrolysis is a thermo-physical process rather than a chemical process. This is significant because no consumable chemical reagents are necessary. Thus, no replacement chemical reagents will be needed for recycling losses that are certain to occur in closed chemical reactors. Second, vapor phase pyrolysis makes extensive use of direct solar thermal energy. This reduces the amount of electrical power required [1].

In the vapor phase pyrolysis process, granulated lunar regolith is vaporized and dissociated. This is done by use of a solar concentrator. The optimum temperatures for vaporization of the granulated lunar regolith is in the range of 2500-3000 K. This is well within the range available for utilization of the direct thermal heating by use of a solar concentrator. After the lunar regolith is vaporized, those vapors are quenched by use of an active cooling mechanism. This process condenses suboxides and will produce free oxygen gas. The gas is then collected and stored. Using this process, oxygen production is approximately 20% of the total processed lunar regolith mass. This process is shown in Figure C.1 [1].

An oxygen plant based on the vapor phase pyrolysis process will only operate during the lunar day. For production of 1.32 kg of oxygen gas per hour from an input of 6.45 kg of lunar regolith per hour, the unit will have a total mass of 565 kg and a electrical power requirement of 12.9 kWe. The mass summary of the unit is shown in Table C.1. If the unit is run ideally at full capacity for 3,800 hrs/yr (43% of a year), the total production will be five metric tons/yr [1]. It should be noted that the mass to production rate ratio is assumed to be constant in these mass approximations.

C.2 HF Acid Leach Process

The HF acid leach process uses low temperature hydro-metallurgical steps to separate the silica content of the lunar regolith from the other metallic oxides by conversions to fluorides and fluorosilicates. This is followed by vaporization of the silica as SiF_4 and the separation of the calcium and structural metals (Ti, Mg, Fe, Al) by a variety of solution, ion-exchange, precipitation, or electrolytic steps. Iron is most easily recovered from solutions by electrowinning. The other metals, except Mg, are recovered by sodium reduction of the corresponding fluorine compounds. Magnesium is recovered by silicon reduction of MgO. Sodium for the corresponding reactions will be recovered through the use of a modified Castner cell. When desired, the metal oxides and silicates will be obtained by hydrolysis of the corresponding fluorine compounds with steam or by ion-exchange methods [2].

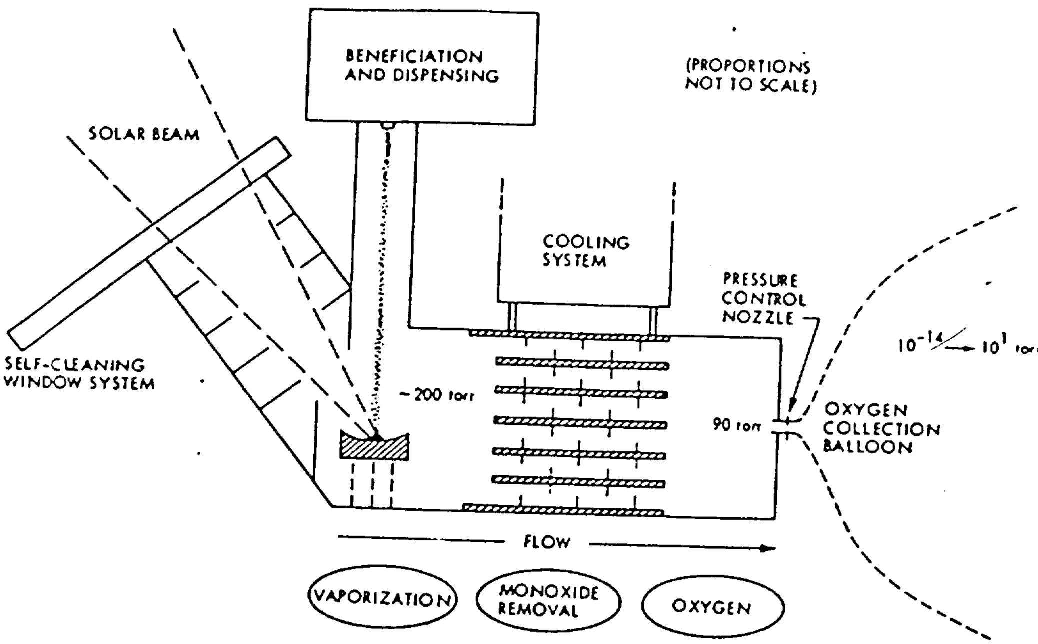

Figure C.1. Extraction of Oxygen from the Lunar Regolith.

Table C.1. Processing Plant Mass Summary.

FACILITIES	MASS (KG)
PROCESSING UNIT	225
SOLAR CONCENTRATOR	75
COOLING SYSTEM	130
COLLECTION SYSTEM	70
SOLAR SHIELDS	15
10% CONTINGENCY	50
PLANT TOTAL	565

A unit with an hourly production of one kg of structural metals and yearly production of seven to eight metric tons per year will have a maximum mass of 320 kg and an electrical power requirement of 6.5 kWe. This does not include the reagent mass estimated to be a mass of 55 kg. The reagent replacement mass will total the initial net unit mass in three to six years. Thus, for a noncontinuous test plant with a conservative lifetime of 10 years, the total mass placed on the lunar surface for a HF acid leach process test plant unit will be 695 kg [2].

C.3 Magnetic Extraction and the Gaseous Carbonyl Process

The lunar regolith is composed of approximately one percent meteoric debris. Meteors fragment into small particles upon impact with the lunar surface. Some of these particles are nearly pure iron and nickel-iron alloys. Analysis of the lunar regolith has determined that the concentration of these nearly pure metallic fragments is between 0.1 to 0.5% of the lunar regolith. Through the use of electromagnets, magnetic extraction of these iron and iron alloy fragments will be rather simple. In fact, magnetic extraction of the meteoric nickel-iron from the lunar regolith is projected to be the simplest method of extracting a metal from known lunar resources [3].

Once the meteoric nickel-iron has been concentrated by use of an electromagnet from the lunar regolith, the gaseous carbonyl process may be used to separate and purify the nickel and iron. In this process, the magnetically extracted nickel-iron will be reacted with carbon monoxide at temperatures of 100 to 200°C and at pressures of 10 to 100 atms. The metals will spontaneously react with the carbon monoxide to produce metallic carbonyls. The carbonyls of the metals are then separated through distillation. At this point, the purified carbonyls are then selectively decomposed at a pressure of one atm and temperatures of 200 to 300°C. Purities of 99.97% are achievable in a single step [4].

The mass and power requirements will be assumed to be similar to that of the HF acid leach unit. The planned unit mass will be 500 kg and the power requirement will be assumed to be six kWe. The planned output is unknown due to the variables in the amount of meteoric iron found on different locations on the surface of the Moon.

C.4 Materials Processing Plants Overview

The total mass of the three material processing units will be 1,760 kg. This includes all consumables necessary for the unit's conservative lifetime of 10 years. During this lifetime a series of remotely operated tests will be performed to determine the viability of each process and the unit design. The initial objective of these plants is to gain engineering experience in materials processing on the lunar surface. The final objective is to gain the technology to enable the creation of a self-sustained lunar industry. Since the FLARE observatory will be assumed to be the only base of operations on the lunar surface, the early experiences with the several material processing units of project FLARE will provide valuable engineering experience for future permanently manned bases on the lunar surface.

References

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2. Waldron, Robert D. and David R. Criswell. "Materials Processing in Space." *Space Industrialization: Volume I.* Ed. Brian O'Leary. Boca Raton, Florida: CRC Press, 1982.
3. Waldron, R. D. and David R. Criswell. "Overview of Methods for Extraterrestrial Materials Processing." *Space Manufacturing III,* Ed. Jerry Grey and Christine Krop. New York: American Institute of Aeronautics and Astronautics, 1979.
4. Lewis, John S. and Ruth A. , *Space Resources: Breaking the Bonds of Earth.* New York: Columbia University Press, 1987.

Appendix D: Communications

D.1 Noise

The channel noise is determined directly by the bandwidth used for transmission. For the proposed lunar base, an initial data transmission rate of 50 Mbps is required. A majority of the information is on video images. Video, however, can be compressed very easily as discussed in the next section. By using compression techniques, we can lower the data transmission rate from 50 Mbps to 250kbps. The 250 kbps translates into a 250 kHz bandwidth requirement. By taking the log to the bandwidth, the channel noise can be determined. In this case, a 250 kHz bandwidth yields a 54 db noise level.

D.2 Compression

Data compression is extremely useful, since it can take a very high data rate requirement and lower it to a level where the design of a telecommunications system becomes more practical. Reduced data rates also improve the efficiency of the system and the bit error performance at the data links.

Uncompressed video images require a large amount data. A frame of 1000 by 1000 bits multiplied by a gray factor of eight requires eight Mbps/frame. Assuming each imager only creates one frame per second, the use of five video cameras on the moon would necessitate an uncompressed data requirement of 40 Mbps. Current image compression techniques can reduce the image by a factor of 500:1 [1]. The assumption of this compression rate would lower the transmission requirements to 80 kbps. This is a significant decrease in the data rate which also allows for a smaller bandwidth.

Data compression is not as easy as video compression. Data is compressed by determining patterns of frequency in the numbers. If all of the observations are random, then it may not be possible to determine a pattern and the data won't be compressed. However, if the observations are the same each time, then they can be compressed, which reduces the transmission rate. The instrument packages on the moon are estimated to produce 52 kbps of data. The robotics telecommunications are estimated to be 120 kbps. This creates an estimated total transmission rate of 175 kbps.

D.3 Fiber Optics

Expected advances in the technology of fiber optics is making this means of communication increasingly attractive. The tactical single-mode fiber optic cable is currently used by the military. This type of cable is much smaller, lighter, and faster than common copper wire cables and can be used in a wide variety of roles. The cable can easily be connected between systems, and current tactical single-mode cable has an outside diameter of two millimeters and is radiation hardened. The radiation hardening treatment would keep the fiber from experiencing an outgassing effect due to a vacuum environment on the moon. Outgassing is the situation where chemical processes within the fiber would begin to occur which would lead to the degradation of the fiber.

Another process being developed by AT&T is that of Eurbidium doping. This doping process of the fiber essentially reduces the amount of refraction within the fiber. This reduction in refraction increases the amount of distance the signal can travel before regeneration.is required. AT&T is also developing a trans-Atlantic cable that would require no signal regeneration. It is hopeful that this type of cable technology, used to span the ocean floor, could be used on the lunar surface in between communication stations.

References

1. Douglas, Robert, *Satellite Communication Technology.* Prentice Hall; Englewood Cliffs, New Jersey. 1988.

Appendix E: Habitation Modules

The first of the two options for the lunar observatory is the inflatable habitat designed by Eagle Engineering. A schematic of the IM is shown in Figure E.1 [1].

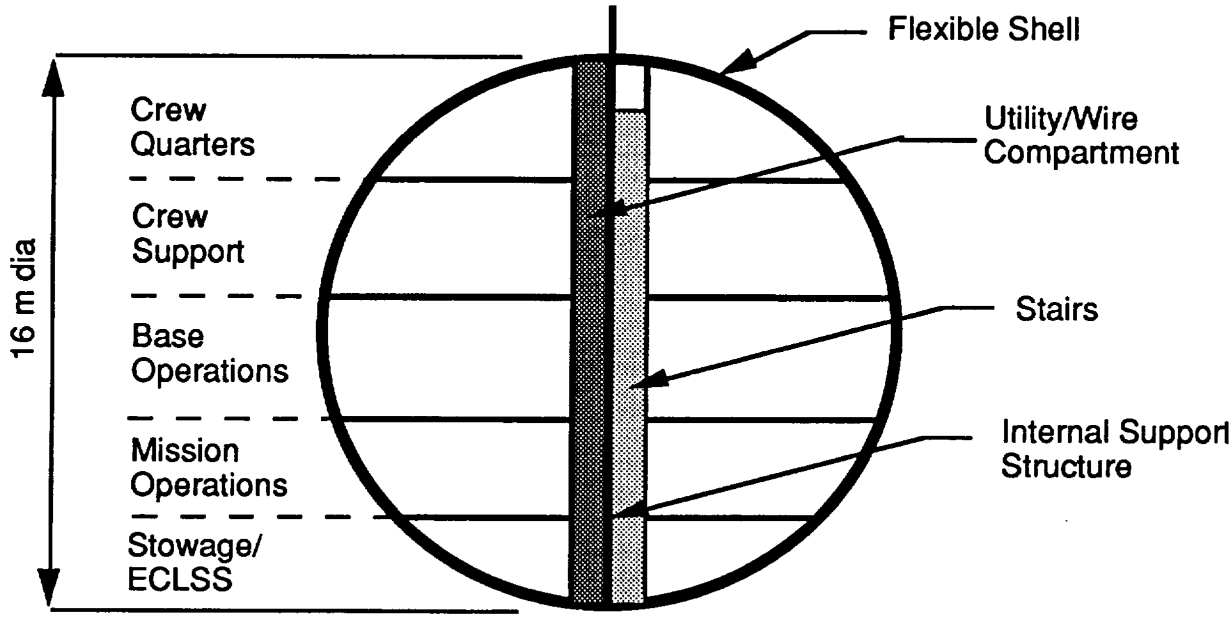

Figure E.1. Schematic of the Inflatable Module.

The inflatable habitat is spherical and sixteen meters in diameter. The IM has four levels in which the crew can live and work, and a fifth level for stowage and the environmental control and life support system (ECLSS). This would provide plenty of habitation space and comfort for four to six crew members -- even during a maximum stay mission of fourteen days.

Two structural systems make up the inflatable habitat. The primary structural system supports the loads from a pressurization of 14.7 psia. It consists of the spherical envelope that surrounds the habitat. The outside of this envelope is made of a strong composite of multi-ply fabric (Kevlar-29) with a strong thermal coating. On the inside of the envelope is a bladder. The secondary structure supports the loads from the crew, furnishings, and equipment. This structure consists of radial floor beams, a spherical rib cage, and columns for support. The levels are separated by lightweight modular floors.

Setting up this type of habitat includes putting it in either a crater or in a purposely made depression and then anchoring it. The setup of the IM will be carried out by robotics. Partial burial of the habitat by lunar regolith is necessary in order to reduce the radiation exposure to crew members. The IM will be inflated to a pressure of 14.7 psia.

References

1. Alfred, John, et al., *Lunar Outpost,* Systems Definition Branch, Advanced Programs Office, Johnson Space Center, 1989.

Appendix F: Delta-V Calculations

St	Input	Name	Output	Unit	Comment
		VLEO	7.6685609	km/s	Velocity in LEO
	398601.2	MUe		km^3/s^2	Gravitational Parameter of Earth
		RLEO	6778.145	km	Radius of LEO
	400	HLEO		km	Altitude of LEO
		AT	195589.07	km	Semi-major axis of transfer ellipse
		VMOON	1.0183044	km/s	Velocity of Moon about Earth
		VP	10.750614	km/s	Vel. at perigee of transfer orbit
		VA	.18956613	km/s	Vel. at apogee of transfer orbit
		DV1	3.0820531	km/s	Delta-v at LEO
		DV2	.80662136	km/s	Delta-v at LLO
		DVTOT	3.8886744	km/s	Total DV from LEO to LLO
		DVIN	.82873825	km/s	Hyperbolic excess speed at Moon
		VO	2.3971234	km/s	Hyperbolic perigee vel. at Moon
	4902.78	MUm		km^3/s^2	Gravitational Parmeter of Moon
		RLLO	1938.09	km	Radius of Low Lunar Orbit
	200	HLLO		km	Altitude of Low Lunar Orbit
		VLLO	1.590502	km/s	Velocity on circular LLO

```
S Rule
* RLEO=HLEO+6378.145
* VLEO = SQRT(MUe/RLEO)
* AT = (RLEO + 384400)/2
* VMOON = SQRT(MUe/384400)
* VP = SQRT(MUe*(2/RLEO-1/AT))
* VA = SQRT(MUe*(2/384400-1/AT))
* DV1 = VP - VLEO
* DVIN = VMOON - VA
* RLLO = HLLO + 1738.09
* VO = SQRT (DVIN^2+2*MUm/RLLO)
* VLLO = SQRT (MUm/RLLO)
* DV2 = VO - VLLO
* DVTOT = DV1 + DV2
```

Appendix G: Legal Analysis of FLARE

Space operations may take place in a physical vacuum, but such operations rarely take place in a political or legal vacuum. Although technical and economic restrictions are usually paramount in mission planning, it is usually a good idea to step back and take a look at how some non-technical restrictions may affect a project. Consequently, the design team at Longhorn Aerospace has decided to evaluate how the present design of the FLARE project would interact with the world of space policy and law.

The primary purpose here is to anticipate the regulations and legal guidelines under which FLARE would have to operate. In most cases, these regulations are not overly restrictive and can be considered as one aspect of the "cost of doing business" in a space environment. However, this idea does not mean that there aren't aspects of FLARE which need closer legal scrutiny.

Overview

Firstly, the legal framework under which the FLARE project will work had to be chosen. This choice was more problematic than might be thought at first. On the one hand, the Outer Space Treaty (OST) is basically the Magna Carta of space policy and law, and almost all nations with an interest in space are signatories to the document. However, another candidate also exists. The Moon Agreement, composed after the OST, is geared specifically toward setting down guidelines for the exploration and use of the moon. Significantly, there are aspects of the Moon Agreement which have proven unpalatable to some, and most major space powers (including the U. S.) still have not signed the document. Still, the fact remains that this agreement is a viable option, and the U. S. may well have become a signatory nation to the Agreement by the time the FLARE project is initiated.

Luckily, the nature of FLARE tends to render moot arguments about which treaty is the most applicable. Overall, the idea of a scientific research expedition is quite compatible with not only the current law of the OST, but also the potential law of the Moon Agreement. For instance, Article I of the OST states that there should be "freedom of investigation in outer space, including the moon . . .," and that the moon should be "free for exploration and use by all States . . . "[1]. Although the legal framework of the Moon Agreement is fairly restrictive toward lunar mining bases, the agreement takes basically the same stand as the OST toward scientific bases, saying that "there shall be freedom of scientific investigation on the moon by all state parties. . ." [2]. Thus it can be seen that either or both treaties can be applied to the FLARE project without significantly shifting the basic framework of the discussion. Consequently, the OST was assumed to be the regime in place for the purpose of setting forth guidelines for FLARE base; however, some consideration was given to certain details of the project wherein the acceptance of the Moon Agreement could entail significant changes.

Of course, it is the specific actions required by these and other relevant treaties which should most concern Lonestar Aerospace, NASA, and the U. S. government. Most of these legal details should have little or no impact on the project. Yet, other aspects certainly require further study and could eventually necessitate a major revision in the mission design. Basically, if the avoidance of legal difficulties is considered to outweigh the avoidance of certain technical difficulties for the overall design concept, a shift in the mission plan may be necessary. The approach for the rest of the analysis was to take a look at all the legal hoops which the FLARE project may have to jump through during and before its operations phase.

Launch Registry

The Convention on Registration of Objects Launched into Outer Space (CROLOS) is fairly specific about the procedure for registering spacecraft flights. Article II of CROLOS states that all objects launched into Earth orbit must be entered into a national registry of such objects [3]. Additionally, the Secretary General of the United Nations must be informed about the parameters of each object entered into the nation's registry. Thus the FLARE project would need to inform the Secretary General of each launch of supplies from Earth, giving the following information in each case:

1) Name of the launching State or States
2) An appropriate designator of the space object or its registration number
3) Date and territory or location of launch
4) Basic Orbital parameters, including:
    i) Nodal period
    ii) inclination

iii) apogee
iv) perigee
5) General function of the space object [3].

Of course, this procedure is standard for all launches, and should not present any problems. Still, it should be noted that the FLARE project will involve numerous such registrations. The large number of registrations results from the fact that the CROLOS also calls for notification of objects which depart from their initial Earth orbit. Therefore, the Secretary General must not only be notified of each earth launch, but also of each major orbital transfer. After the Earth launch of elements of FLARE, the elements must be transferred from the Space Station orbit to the parking orbit of the OTV. Multiple trips by a space tug may be required to fully load the OTV, with each trip having to be registered. Next, the transfer of the OTV from Earth orbit to LLO, the orbital parameters of LLO, the transfer of the lander to the lunar surface, and the return of the OTV must all be registered. Admittedly, all the parameters during the OTV phase of operations could be included in one giant flight plan for the purpose of registration. However, the fact that six OTV trips are required by FLARE means that the Secretary General will still receive a fair amount of correspondence from NASA during the project.

Also of note is the status of registering nuclear payloads, such as the TOPAZ nuclear reactors to be used by the base. Neither OST nor CROLOS has any specific mention of informing the Secretary General about nuclear payloads. The closest that either treaty comes to doing so is CROLOS's request for a general function of each item registered. However, if the U. S. ever became a signatory to the Moon Agreement, FLARE would then be required to "notify [the Secretary General] in advance of all placements . . . of radioactive materials on the moon and the purpose of such placements" [2].

Liability Considerations

At this point, it would be a good idea to step aside and take a look at the subject of liability for damages under current space law. Space activities are not no-risk propositions, and FLARE's practice of incorporating large amounts of nuclear technology in its design means that, in the event of an accident, the list of damage claims could be quite large. The applicable law, here, is covered by the Convention on International Liability for Damage caused by Space

Objects (CILDSO), which states that a "launching state shall be absolutely liable to pay compensation for damages caused by its space object . . ." [4]. Normally, liability for damages is just another cost of doing business in space. However, the example of the breakup of the TOPAZ-powered Soviet satellite Kosmos 954 in 1978 shows that the launching and orbiting of nuclear packages should be of special concern in mission planning. In this case, the orbit of the satellite decayed, and mildly radioactive debris were strewn across a sparsely inhabited section of Canada. Even though few people lived in the debris path, the Soviet Union was liable for damages in the amount of three million dollars Canadian [5]. This occurrence indicates that special care should be taken in the Earth launch and orbit of any packages containing nuclear reactors.

Another serious concern is the potential of the OTV for causing damage. Nuclear-electric propulsion is still an unproven technology, and the mercury used as fuel could be very damaging if it ever spread into the Earth's environment. Of course, these considerations have already entered the mission design in the form of the nuclear-safe parking orbit and the requirement that the thrust vector of atomically vaporized mercury never be pointed toward the Earth. However, accidents happen, and an inadvertent misalignment of thrusters could send a small cloud of radioactive mercury on a collision path with the Earth. Although the effects of such an event are unknown, there is no doubt that there would be widespread claims against the U. S. government in the wake of such an accident.

In Article XI, CILDSO outlines the procedure which would then be followed. Firstly, claims for compensation for damage could be presented to the U. S. through diplomatic channels or through the offices of the Secretary General. On the other hand, claims could instead be presented in the federal courts of the U. S. [4]. However, the compensation for damages would be guided by international law in either case. Moreover, CILDSO provides for measures to force the settlement of claims fairly rapidly. If diplomatic negotiations cannot settle a claim within a year of the filing of the claim, CILDSO states that a claims commission must be created. This commission, in turn must render a verdict within a year of when it receives jurisdiction [4].

Because there is no guarantee that the U. S. government could get the claims dismissed, the nuclear aspects of FLARE should be subjected to further technical and legal research before the TOPAZ generators and the nuclear-electric OTV are locked irrevocably into the mission design. Specifically, since the environmental effects of a mercury cloud impact are unknown at this time, a study should be done to estimate the likely consequences from such an accident. If the risks involved with nuclear-electric propulsion are established to be small, the project may proceed unchanged. If the potential damage is large and widespread, the OTV may have to be redesigned with even greater safety margins or dropped altogether from the design concept.

Environmental Questions on the Moon

The Earth is not the only place where space law protects the environment. The OST calls for exploration of celestial bodies to be conducted "so as to avoid their harmful contamination . . ."[1]. While this statement is somewhat vague and does not seem to offer stringent protection for the lunar environment, the ratification of the Moon Agreement could impose some strict guidelines on the FLARE project for the protection of the lunar environment. Indeed, the language of the Agreement is fairly strong on this point:

> In exploring and using the moon, state parties shall take measures to prevent the disruption of the existing balance of its environment whether by introducing adverse changes in that environment, by its harmful contamination through the introduction of extra-terrestrial matter, or otherwise [2].

Although the moon is commonly seen as a lifeless planet, swimming in interstellar vacuum, this view is not wholly accurate. The moon does possess an atmosphere--albeit a thin and tenuous one. It has been estimated that each Apollo mission released gases that temporarily *doubled* the lunar atmosphere. Although the thickened atmosphere quickly decayed back to its normal state, Vondrak has estimated that even modest lunar exploration--such as is envisioned by FLARE--could release gases which would normally stay locked in the lunar surface [6]. Thus, gases of natural origin would shortly become only trace components in the lunar atmosphere [6]. Such an occurrence would directly contradict the Moon Agreement, and could lead to legal problems. Still, unless a moratorium on lunar development is imposed, a shift in the composition of the lunar atmosphere may be inevitable anyway. Consequently, it is unreasonable to severely curtail FLARE solely on the basis of this problem. Although, the language of the Moon Agreement says that measures should be taken to prevent disruption, this does not mean that disruption must *absolutely* be prevented.

If the U. S. ratifies the Moon Agreement, FLARE can negate any legal problems with the environmental clause by creating special operations procedures to minimize disruptions of the atmosphere. By informing the Secretary General of these measures, as is also required by the Moon Agreement, the FLARE project should be able to proceed with a minimum of problems. A necessary first step for the creation of these protective measures would be to conduct some kind of environmental impact study. In this study, FLARE should consider sending its environmental observatory down to the lunar surface long before the project actually begins. In this manner, the makeup of the *natural* lunar atmosphere can be better established. This model can then, in turn, be used to see how the FLARE base might affect the lunar atmosphere and also to establish what procedural measures can be used to limit disruption of the atmosphere.

Materials Processing

The prototype materials processing plants will probably be the main factor responsible for unlocking gases from the lunar surface. Additional questions remain, however, about these plants. Under the Outer Space Treaty, there are no real barriers for setting up materials processing operations. The only requirements are that the Secretary General must be informed of the nature and conduct of all such activities and that the relatively mild environmental considerations that have already been mentioned must be considered.

However, the possibility of Moon Agreement ratification again presents some tougher questions for FLARE. Article 11 of the Moon Agreement and its "common heritage of mankind" (CHM) clause has probably been the main reason that the U. S. has not ratified the document. The principles of CHM are that space, and the bodies within space, are the common heritage of all mankind--including the generations of man still to come. Thus, no one section of humanity, such as the U. S., can appropriate part of the moon for its exclusive use. Specifically, exploitation is limited by the statement that

> Neither the surface nor the subsurface of the of the moon, nor any part thereof or natural resources in place, shall become the property of any state, . . ., national organization, or . . . any natural person [2].

However, other sections of the Moon Agreement mitigate this difficulty--at least in the case of the FLARE base. Because FLARE is a scientific research expedition and therefore focuses on exploration, rather than exploitation, it is allowed considerable leeway in using the natural resources of the moon. In Article six , the Moon Agreement gives states the right to "use mineral and other substances of the moon in quantities appropriate for the support of their missions" [2]. In other words, as long as the production from the materials processing plants is small and the results go toward improving the bases infrastructure (*i.e.*, using the oxygen plant as a backup oxygen supply), the Moon Agreement presents no real problems for FLARE. If FLARE had intended to process the lunar regolith in a profit-oriented business, on the other hand, the Agreement would have certainly come into play in a more active fashion. For example, some have interpreted the "common heritage of mankind" clause to mean that profits from lunar exploitation should be distributed to all mankind [7]. This practice would quickly make such a development project impractical, to say the least. But as Galloway has said, "freedom of scientific investigation and use of natural resources . . . are not affected by the guidelines created for regulating the exploitation of natural resources" [7].

<u>Recommendations</u>

As long as the Outer Space Treaty remains the legal framework for the United States in lunar development, the FLARE project can anticipate no major legal difficulties in its operations. All the legal requirements should be the standard ones required by all space flights. The only novel action to be recommended in this case is to conduct a thorough study on the risks of, and potential liability for, nuclear accidents in the course of the FLARE project. Although at present, the risks are judged to be acceptable, this action should be taken as soon as possible, so as to limit the possibility of getting locked into a design with a technology that later proves unwise in light of a more thorough study.

On the other hand, if the U. S. approves the Moon Agreement before the commencement of the project, additional actions may be necessary. FLARE would then have to take the additional actions of reporting the TOPAZ placements on the moon and also of reporting the measures being taken to limit disruption of the lunar environment. Additionally, it is recommended--although not required--that an environmental impact study be conducted to determine how the FLARE project will affect the lunar atmosphere.

References

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